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(NASA-CR-192456) PRELIMINARY BASE HEATING ENVIRONMENTS FOR A GENERALIZED ALS LO2/LH2 LAUNCH VEHICLE, APPENDIX 1 AND 2 (Remtech) 352 p

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REMTECH TECHNICAL NOTE

TITLE: PRELIMINARY BASE HEATING ENVIRONMENTS FOR A GENERALIZED ALS LO_2/LH_2 LAUNCH VEHICLE

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CONTRACT: NAS8-38141

PREPARED FOR: NASA/MSFC Induced Environment Branch ED-33

Introduction

A secondary objective of contract NAS8-38141 is for REMTECH to provide base heating assessments, as required, to support Advanced Launch System (ALS) preliminary launch vehicle and propulsion system design studies. The ALS propulsion systems integration working group meeting (No. 3) recently completed in San Diego, California, focused attention on the need for base heating environment determination to provide preliminary requirements for LO_2/LH_2 propulsion systems currently being considered for ALS. REMTECH was requested to provide these environments for a range of possible propellant mixture and nozzle area ratios.

Base heating environments can only be determined as a function of altitude when the engine operating conditions and vehicle base region geometry (engine arrangement) are known. If time dependent environments are needed to assess thermal loads, a trajectory must also be provided. These parameters are not fixed at this time since the ALS configurations and propulsion operating conditions are varied and continue to be studied by Phase B contractors. Therefore, for this study, a generalized LO_2/LH_2 system was selected by REMTECH along with a vehicle configuration consisting of a seven engine-booster and a three-engine core. MSFC provided guidance for the selection.

REMTECH also selected a limited number of body points on the booster and core vehicles and engines for the environment estimates. Environments at these locations are representative of maximum heating conditions in the base region

and are provided as a function of altitude only. Guidelines and assumptions for this assessment, methodology for determining the environments, and preliminary results are provided in this technical note. Refinements in the environments will be provided as the ALS design matures.

Propulsion System

The first step in the analysis was the selection of the propulsion system data and configuration information germane to base heating.

For the propulsion system, this includes chamber conditions, throat or nozzle exit area, and length from gimbal point to the nozzle exit. The conditions selected are:

Propellants: LO_2/LH_2
Chamber Pressure: 2250 psia
Mixture Ratio: 5.5 and 6.0
Throat Diameter: 12.9 inches
Nozzle Area Ratio: 35, 45, and 60
Length from Gimbal to Nozzle Exit: 150" for $A/A_* = 35$
185" for $A/A_* = 45$
240" for $A/A_* = 60$

Propulsion exhaust products and properties at different expansion ratios were determined for this system at both mixture ratios by the CEC code, Ref. 1.

Configuration

A mated configuration consisting of a three-engine core vehicle and seven-engine booster was selected as shown schematically in Fig. 1. Both the booster and core elements were assumed to be 360 inches in diameter for the main propellant tanks with the main thrust frame for both elements in the same plane. Both elements were assumed to have a cylindrical aft skirt attached to the thrust frame, extending aft approximately 96 inches.

For the booster, the seven engines were arranged as shown in Fig. 2 with six engines equally spaced around a center engine. Centerline-to-centerline spacing between engines was approximately 120 inches. The aft skirt extends midway down the nozzle for the 35/1 area ratio nozzle as depicted in Fig. 2. A flat surface

heat shield was assumed to connect all seven nozzles in a plane parallel to the thrust frame and approximately 90 inches aft.

The core vehicle geometry and engine arrangement selected for the study is shown in Fig. 3. The three engines are arranged in an equilateral triangle configuration on a nozzle centerline circle diameter of approximately 100 inches. Aft skirt and base heat shield geometry were assumed to be the same as the booster. One engine of the core vehicle was positioned in closest proximity to the booster such that the remaining two engines are equal distance from booster as shown in the lower schematic of Fig. 1.

Assumptions and Study Guidelines

For this study, vehicle forebody geometry and freestream flow effects on the plumes and base heating environments were ignored since a trajectory was not provided. Also, the small differences in properties between the 5.5 and 6.0 mixture ratio did not warrant separate studies; therefore, all computations were based upon the 6.0 mixture ratio thermodynamic and transport data. In addition, the flowfield changes and resultant environment shifts due to 9 degree gimbaling on all engines were assumed to be relatively small and within the conservatism dictated by the methodology. This simplification produced study results for all engines firing aft with parallel burn assumed for both the booster and core.

Additional assumptions included all engines firing at full thrust, turbopump exhaust disposal and burning occurring in the nozzle, and no mass or energy additions to the base flow from flow deflectors, vents, etc. in the aft skirt or heat shield.

Guidelines for the study (provided by MSFC) were to determine environments at critical base locations at several altitudes from sea level to 100,000 feet. Three cases were to be investigated.

Case 1: All booster and core engines had nozzles with $A/A_* = 35$

Case 2: All booster and core engines had nozzles with $A/A_* = 45$

Case 3: All booster and core engines had nozzles with $A/A_* = 60$

In addition, a hybrid case with booster engine at $A/A_* = 35$ and core engines at $A/A_* = 60$ was also of interest. Engine mixture ratio variation was assumed to

be within the overall accuracy of the study, as mentioned previously, and was not considered.

Body Point Selection

REMTECH selected seven locations on the booster and nine locations on the core as points for which environments would be determined. These locations are shown in Figures 4 and 5 for the booster and core respectively. As much as possible, the locations were selected where maximum heating to the engine nozzle, base heat shield, and aft skirt would occur. Obviously, aft facing locations on the nozzle and aft skirt trailing edge are likely to receive maximum radiation since they have unobstructed views of several plumes. Maximum convection may also occur on the nozzle; however, it will be much later in flight; historically at altitudes around 100,000 feet when peak recircultion occurs. Typical locations of interest on the base heat shield for convective heating were selected to demonstrate peak interior heating, vent plane heating, and average exterior (to the engine circle) heating. Body point notation is a two letter plus numerical designation; the first letter refers to either booster or core and the second letter refers to nozzle, base heat shield, or skirt: e.g. - BN2 is the second point on the booster nozzle.

Prediction Methodology

The base heating environment consists of radiative and convective components. Infrared radiation from the rocket exhaust plumes varies strongly with surface position as views of the plumes from the surface and shading by other surfaces change. In contrast, the convective heating environment, resulting from reversed plume boundary layer gases, is essentially constant over relatively wide areas. Both heating modes are basically a function of altitude with flight-time effects also entering through variations in engine chamber pressure. In this section, the radiative and convective components prediction methodology will be discussed independently.

Plume Radiation Methodology

The methodology was based on scaling of existing SSME plume property data where this was suitable. In other cases, extrapolations of these results were made using simplified plume models to cover the full range of nozzle area ratios and base configurations considered.

Because of the availability of SSME plumes for scaling, the radiation predictions for the ALS were all made using a mixture ratio ($L0_2/LH_2$) of 6.0. This mixture ratio is expected to produce higher radiation than a mixture ratio of 5.5 because it produces higher temperatures, pressures (for a given area ratio), and water-vapor mole fractions. Although the lower mixture ratio has a greater fraction of LH_2 for afterburning, this is not expected to offset the greater radiation potential of the higher mixture ratio.

One-dimensional, equilibrium predictions indicated that the area ratio 60 nozzle and the SSME have nearly identical exit pressures, so the shock structure is expected to be similar. However, because of the lower chamber pressure of the ALS engines, the plume temperature will be slightly higher when expanded to any pressure. Therfore, the SSME plumes at sea level, 20 kft and 40 kft were scaled to increase temperature by 6 percent and lengths by 10 percent to simulate the area ratio 60 ALS engine. Variations with altitude used the adjustment functions [2] normally applied for the Space Shuttle.

Extrapolation to other area ratios were made using idealized inviscid plumes with the properties from 1-D equilibrium predictions. These were combined with judgements of afterburning effects based on the scaled SSME plumes for the area ratio 60 ALS nozzles with the core of the plume removed to emphasize the afterburning.

Convective Base Heating Methodology

Convective heating from recirculated plume gases is not determined by a rigorous computational procedure or computer code, but relies on judicious scaling and application of existing flight and model data. For this study, which considered only LO_2/LH_2 propulsion systems, the prediction methodology relied heavily on Saturn I/S-IV stage, Saturn V/S-II stage, and Shuttle Orbiter flight data. Model data trends for clusters of engines and various chamber conditions were also used in the analysis.

Data from the LO_2/LH_2 database were arranged to show the effects of chamber pressure, nozzle area ratio, and base region vent area (nozzle spacing) on maximum heating. From these trend curves, individual scaling factors were determined to account for each parameter of the ALS engine performance or base geometry which differed from a known, measured enironment; in this case, convective heating to the Shuttle SSME nozzle. Using these factors in series it was possible to adjust the Shuttle data to conditions under investigation for the ALS.

Additional corrections to the Shuttle adjusted environment were made from

generalized trends evident in the global base heating database possessed by REMTECH. For example, axial variations in heating along the nozzle and radial variations in heating across the heat shield were available from distribution curves extracted from the database and normalized to nozzle exit diameter or engine spacing. Effects of heat shield position and aft skirt length were also determined from the database trends.

No attempt was made, at this time, to define the plumes at different altitudes and visually assess plume interactions. Instead, other flight data with a variety of nozzle spacings were utilized to estimate the onset of recirculation, the altitude of maximum convective heating, and the altitude of initial choked flow in the base. These three critical altitudes were determined for both the booster and core engine spacing at each of the three nozzle sizes under consideration. In general, the onset of recirculation varied from 40,000 to 80,000 feet depending on the configuration and nozzle area ratio. Peak convection usually occurred around 100,000 feet and choked base flow is established above 150,000 feet.

Curves of cold wall convective heating rate were determined as a function of altitude for each of the booster and core body points by smoothing through the estimates at the three critical altitudes. A single value of base gas recovery temperature applicable to all base region surfaces was also estimated from correlations with nozzle exit Reynolds number and previous flight experience. Maximum gas temperature in either the booster or core base is expected to be about 2900°R or approximately 45 percent of the chamber temperature.

Results

Radiation and convective base heating environments have been determined separately and are presented in tabular form as a function of altitude. Separate environments were determined for the booster and core body point for each of the three LO_2/LH_2 nozzle area ratios of interest.

The radiation results for the core and booster vehicles are presented in Tables 1 and 2 for the base heat shield CB1/3 and BB1/3, the nozzle lip CN1/2 and BN1/2, and the skirt around the base CS1/2 and BS1/2. Characteristics of the radiation to each component will be discussed in the following paragraphs.

Experience on the Space Shuttle heat shield has indicated that the higher rates occur outside the engine circle because of shading by the nozzles in the center of the cluster. This effect is not expected to be as pronounced on the ALS Core vehicle analyzed because of the greater relative spacing between engines. As the engine area ratio is reduced, the base has a better view of the plumes and the

rates are predicted to increase slightly. For the configuration chosen for analysis, points viewing the plume from the base aspect are most affected by the plume size and afterburning rather than the shock structure close to the nozzle. The high area-ratio nozzles are relatively far from the base and the shock structure will be be relatively shaded from the base. Radiation for the ALS booster configuration studied, increases slightly relative to the core vehicle because of the increased number of plumes and the tighter cluster.

The aft facing nozzle lip (Point N1) is sensitive to all aspects of the plume radiation since it has a good view of all of the plumes. It will tend to increase as a function of the temperature and pressure of the adjacent gas at the nozzle exit and with the reduction in distance to adjacent plumes. In the case of high area-ratio nozzles, the adjacent gas is relatively cool and low pressure, but the Mach disk occuring a short distance downstream is an intense radiation source. The results indicate a reduction with decreasing area ratio because of the weakening of the Mach disk and an increase in distance to adjacent plumes because of the smaller exit size. In the case of the ALS booster vehicle, the rates increase slightly because of the closer engine spacing.

The lateral facing surface at the nozzle lip (point N2) sees only the adjacent plume and has a better view of the region near the exit than the downstream afterburning region. As a result it is sensitive to engine spacing and exit conditions. It is likely to decrease rapidly as shocks weaken with increasing altitude.

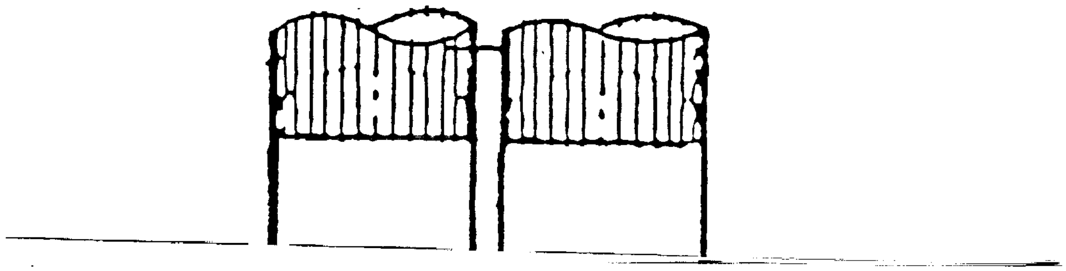
Results for the two skirt points (S1 and S2) on the core vehicle indicated no significant sensitivity to area ratio, so the results show no area ratio effect. However, the booster skirt is closer to the plumes than for the core vehicle configuration analyzed, and the outboard engine spacing is closer. As a result, the booster rates are higher and show a more significant shading effect as larger nozzles are used.

Cold wall convective heating rate and heat transfer coefficient plus base gas recovery temperature are presented in Tables 3 through 8. Maximum convection occurs on the center engine nozzle exit on the booster with the smallest nozzle ($A/A_* = 35$) as expected since the highest density reversed flow and more intense recirculation are indicated. In general, the core vehicle convective heating is less severe that the booster and has a shorter exposure to the recirculated flow. Increasing the nozzle exit ratio generally reduces the heating, although the reduction is moderated by slightly earlier (lower altitude) recirculation since the nozzle exits are in closer proximity. Base interior heating is more severe than peripheral heating as expected. Any significant change in geometry, engine arrangement, or chamber condition could dramatically alter these results.

Reference

1. Svehla, Roger A. and McBride, Bonnie J., "FORTRAN IV Computer Program for Calculation of Thermodynamics and Transport Properties of Complex Chemical Systems," NASA TN D-7056, Jan. 1973.

2. J.E. Reardon and Y.C. Lee, "Space Shuttle Main Engine Plume Radiation Model," REMTECH RTR 014-6, December 1978.



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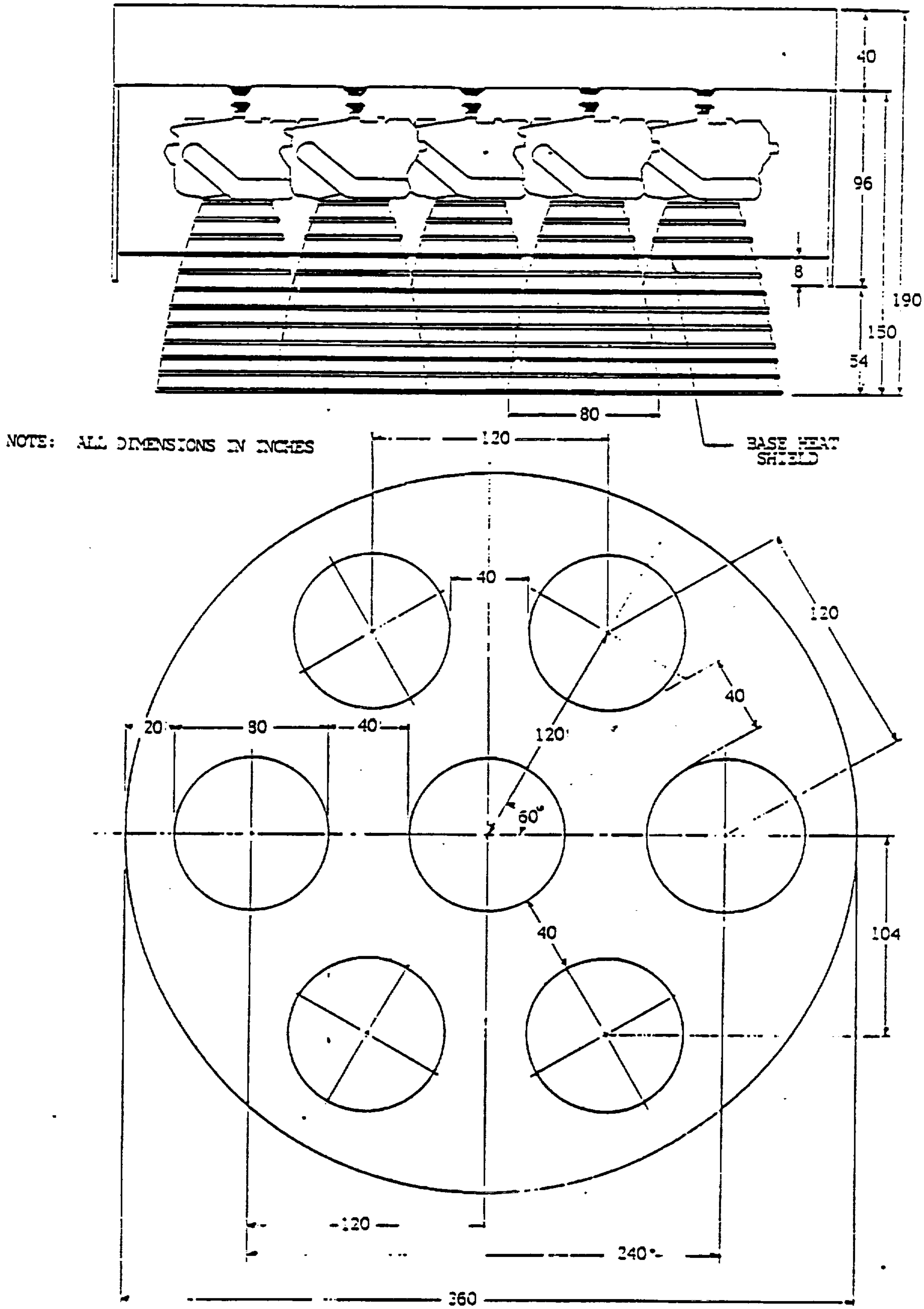

Figure 2: ALS Booster Base Configuration (Assumed)

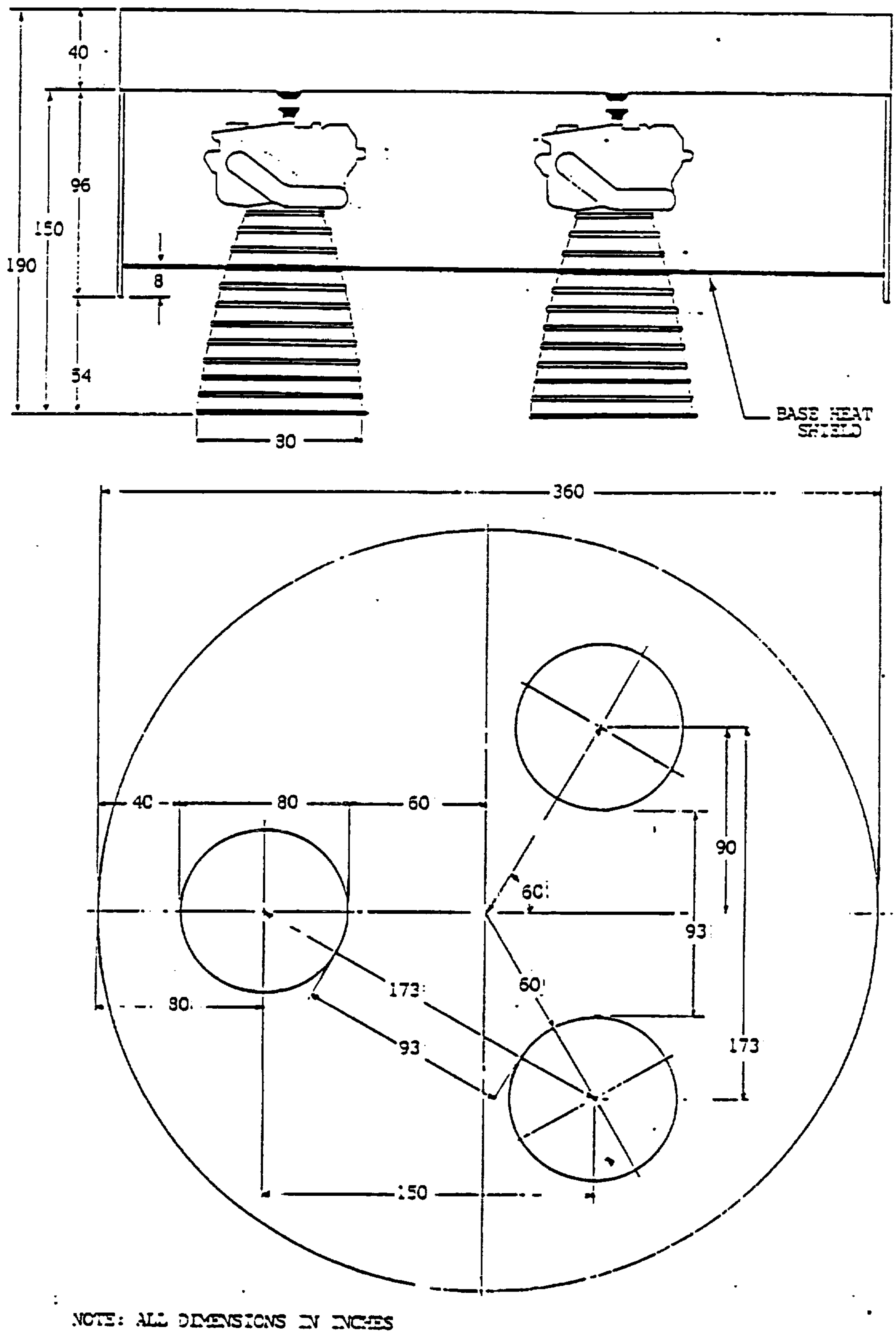

Figure 3: ALS Core Base Configuration (Assumed)

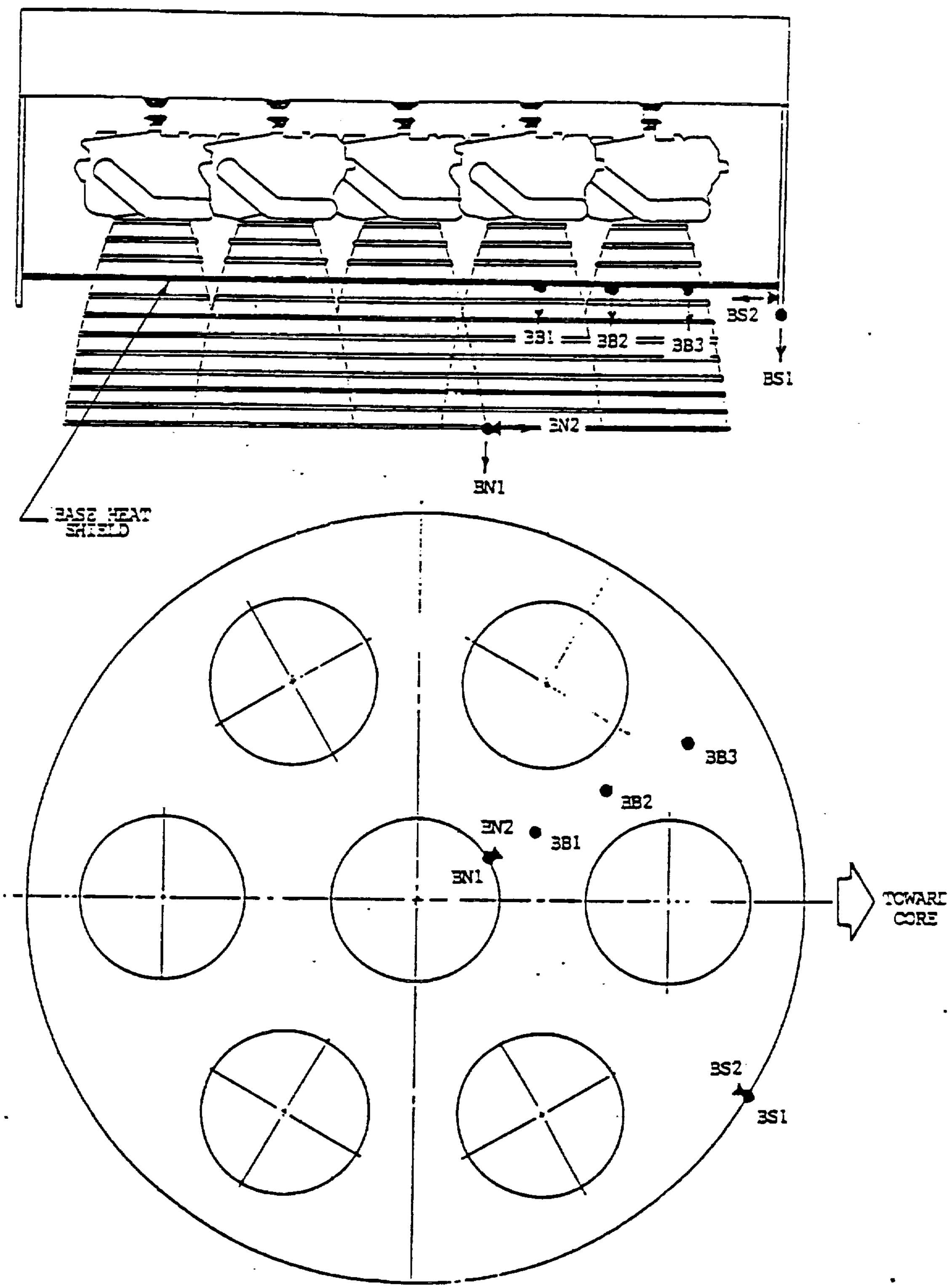

Figure 4: Booster Locations Selected for Base Heating Analysis

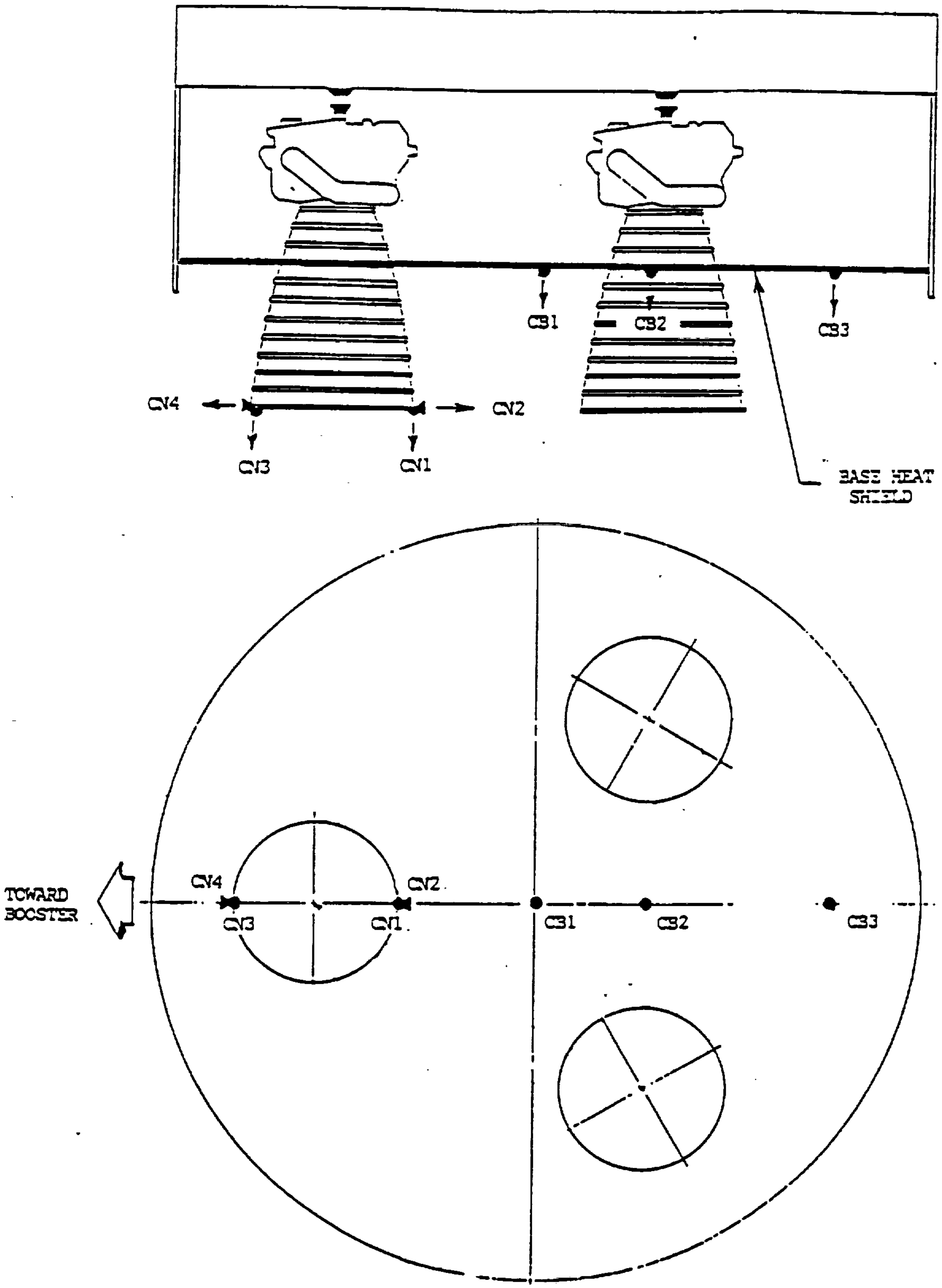

Figure 5: Core Locations Selected for Base Heating Analysis

Table 1: ALS Booster Incident Plume Radiation Rates (Btu/sq-ft-sec)

ALS BOOSTER

INCIDENT PLUME RADIATION RATES (Btu/sq-ft-sec)

35 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT)				
	0	10	30	50	100
BB1	7.0	6.6	4.8	2.5	1.3
BB2	7.0	6.6	4.8	2.5	1.3
BB3	7.9	7.4	4.5	2.3	0.7
BN1	21.0	18.0	11.7	9.2	7.0
BN2	10.5	9.0	4.5	3.5	3.0
BS1	13.4	12.0	6.2	4.2	1.6
BS2	3.7	3.3	1.5	0.8	0.6

45 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT)				
	0	10	30	50	100
BB1	6.4	6.1	4.1	2.2	1.3
BB2	6.8	6.4	3.7	1.9	1.1
BB3	6.8	6.4	3.7	1.9	1.1
BN1	28.0	24.0	12.0	8.5	6.0
BN2	14.0	12.5	4.9	3.5	3.0
BS1	10.8	9.8	5.2	3.5	1.4
BS2	3.0	2.7	1.2	0.7	0.6

60 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT)				
	0	10	30	50	100
BB1	5.1	4.6	2.4	1.4	1.0
BB2	4.1	3.8	2.3	1.3	1.0
BB3	4.0	3.7	2.2	1.3	1.0
BN1	40.0	35.0	15.2	12.1	8.6
BN2	20.0	16.6	6.0	5.0	4.3
BS1	5.7	5.3	3.2	2.0	1.0
BS2	1.7	1.5	0.6	0.5	0.5

Table 2: ALS Core Incident Plume Radiation Rates (Btu/sq-ft-sec)

ALS CORE

INCIDENT PLUME RADIATION RATES (Btu/sq-ft-sec)

35 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT) 0	10	30	50	100
CB1	5.8	5.5	4.0	2.1	1.1
CB2	5.8	5.5	4.0	2.1	1.1
CB3	7.2	6.7	4.1	2.1	0.6
CN1	20.0	15.0	7.6	6.2	5.0
CN2	6.0	4.3	2.3	1.7	1.6
CN3	21.0	16.0	7.4	6.5	5.3
CN4	9.0	7.1	3.7	2.6	2.5
CS1	8.4	7.6	3.9	2.0	1.0
CS2	3.0	2.7	1.2	0.7	0.5
45 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT) 0	10	30	50	100
CB1	5.6	5.3	3.6	1.9	1.1
CB2	5.6	5.3	3.6	1.9	1.1
CB3	6.8	6.4	3.7	1.9	0.5
CN1	25.0	18.0	8.0	6.2	5.0
CN2	8.0	6.0	2.4	1.7	1.6
CN3	27.0	21.0	10.2	7.3	5.9
CN4	11.5	9.5	3.7	2.6	2.4
CS1	8.4	7.6	3.9	2.0	1.0
CS2	3.0	2.7	1.2	0.7	0.5
60 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT) 0	10	30	50	100
CB1	5.3	5.0	3.4	1.8	1.0
CB2	5.3	5.0	3.4	1.8	1.0
CB3	6.5	6.1	3.5	1.8	0.5
CN1	30.0	27.0	10.0	6.2	4.0
CN2	10.0	8.2	2.6	1.7	1.5
CS1	8.4	7.6	3.9	2.0	1.0
CS2	3.0	2.7	1.2	0.7	0.5

Table 3

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

++++++++++++++++++++++++
BOOSTER
++++++++++++++++++++++++

A/A* =35.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: BN1 Qc	hc	Body Pt: BN2 Qc	hc	Body Pt: BS1 Qc	hc	Body Pt: BS2 Qc	hc	Body Pt: BB1 Qc	hc	Body Pt: BB2 Qc	hc	Body Pt: BB3 Qc	hc
58	1605	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
60	1660	0.5	.443E-03	0.5	.433E-03	0.4	.295E-03	0.3	.264E-03	0.4	.316E-03	0.3	.264E-03	0.3	.232E-03
70	1910	4.2	.291E-02	3.5	.243E-02	3.5	.241E-02	3.0	.208E-02	3.5	.243E-02	2.5	.174E-02	1.9	.133E-02
80	2125	9.7	.584E-02	8.6	.519E-02	8.0	.483E-02	7.0	.423E-02	7.4	.446E-02	5.9	.355E-02	4.1	.247E-02
90	2310	17.0	.920E-02	16.0	.867E-02	13.5	.732E-02	11.4	.618E-02	11.9	.646E-02	8.1	.435E-02	5.5	.297E-02
100	2465	18.6	.927E-02	17.6	.878E-02	14.9	.751E-02	12.4	.620E-02	13.2	.658E-02	8.4	.422E-02	5.5	.276E-02
110	2600	16.8	.787E-02	16.3	.763E-02	13.7	.641E-02	11.5	.537E-02	12.2	.569E-02	7.7	.361E-02	4.9	.230E-02
120	2710	13.9	.619E-02	13.5	.598E-02	11.1	.491E-02	8.9	.397E-02	9.5	.420E-02	6.0	.268E-02	3.9	.175E-02
130	2795	10.6	.455E-02	10.0	.430E-02	8.3	.354E-02	6.9	.294E-02	7.2	.309E-02	4.7	.201E-02	3.2	.135E-02
140	2855	8.4	.349E-02	8.0	.336E-02	6.7	.281E-02	5.8	.243E-02	6.0	.250E-02	3.9	.162E-02	2.6	.107E-02
150	2900	7.6	.310E-02	7.2	.295E-02	6.0	.246E-02	5.1	.209E-02	5.3	.218E-02	3.4	.141E-02	2.3	.924E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 4

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

BOOSTER

A/A* =45.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

Alt ft 10**3	Tr R	Body Pt: BN1 Qc	Body Pt: BN1 hc	Body Pt: BN2 Qc	Body Pt: BN2 hc	Body Pt: BS1 Qc	Body Pt: BS1 hc	Body Pt: BS2 Qc	Body Pt: BS2 hc	Body Pt: BB1 Qc	Body Pt: BB1 hc	Body Pt: BB2 Qc	Body Pt: BB2 hc	Body Pt: BB3 Qc	Body Pt: BB3 hc
45	1270	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
50	1385	0.9	.945E-03	0.8	.877E-03	0.7	.712E-03	0.5	.575E-03	0.5	.542E-03	0.4	.371E-03	0.2	.205E-03
60	1685	3.0	.250E-02	2.9	.243E-02	2.7	.220E-02	2.2	.184E-02	2.4	.200E-02	1.4	.115E-02	0.9	.740E-03
70	1910	6.3	.431E-02	6.0	.414E-02	5.3	.365E-02	4.3	.296E-02	4.6	.317E-02	2.9	.199E-02	1.8	.127E-02
80	2125	11.0	.661E-02	10.2	.606E-02	8.8	.520E-02	7.0	.415E-02	7.6	.450E-02	4.9	.290E-02	3.0	.183E-02
90	2310	13.9	.744E-02	13.2	.708E-02	11.1	.601E-02	9.3	.504E-02	9.8	.526E-02	6.1	.332E-02	4.1	.224E-02
95	2390	14.1	.725E-02	13.4	.691E-02	11.3	.583E-02	9.6	.498E-02	10.0	.521E-02	6.3	.328E-02	4.3	.222E-02
100	2465	14.0	.698E-02	13.3	.664E-02	11.2	.560E-02	9.5	.473E-02	10.0	.498E-02	6.2	.312E-02	4.2	.209E-02
110	2600	12.9	.604E-02	12.2	.569E-02	10.2	.477E-02	8.7	.404E-02	9.1	.426E-02	5.5	.259E-02	3.5	.166E-02
120	2710	10.1	.450E-02	9.6	.425E-02	8.4	.374E-02	7.1	.315E-02	7.5	.333E-02	4.4	.194E-02	3.0	.133E-02
130	2795	8.0	.344E-02	7.7	.329E-02	6.9	.296E-02	5.8	.250E-02	6.1	.262E-02	3.7	.157E-02	2.5	.109E-02
140	2855	6.7	.281E-02	6.5	.270E-02	5.8	.243E-02	4.8	.202E-02	5.1	.214E-02	3.2	.134E-02	2.2	.923E-03
150	2900	6.0	.246E-02	5.7	.235E-02	5.0	.207E-02	4.2	.171E-02	4.4	.182E-02	2.8	.116E-02	2.0	.800E-03
160	2900	5.6	.231E-02	5.4	.221E-02	4.6	.187E-02	3.8	.156E-02	4.1	.166E-02	2.6	.105E-02	1.8	.720E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 5

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

++++++++++++++++++++++++
BOOSTER
++++++++++++++++++++++++

A/A* =60.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

Alt ft 10**3	Tr R	Body Pt: BN1 Qc	hc	Body Pt: BN2 Qc	hc	Body Pt: BS1 Qc	hc	Body Pt: BS2 Qc	hc	Body Pt: BB1 Qc	hc	Body Pt: BB2 Qc	hc	Body Pt: BB3 Qc	hc
40	1095	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
45	1240	0.7	.897E-03	0.6	.773E-03	0.5	.659E-03	0.4	.513E-03	0.4	.515E-03	0.2	.269E-03	0.2	.204E-03
50	1385	1.4	.157E-02	1.3	.140E-02	1.2	.132E-02	0.9	.977E-03	1.1	.119E-02	0.5	.566E-03	0.4	.388E-03
60	1660	3.3	.272E-02	3.1	.261E-02	2.8	.235E-02	2.1	.176E-02	2.5	.206E-02	1.3	.106E-02	0.8	.695E-03
70	1910	5.6	.385E-02	5.4	.372E-02	4.7	.326E-02	3.6	.247E-02	4.1	.280E-02	2.3	.157E-02	1.5	.106E-02
80	2125	8.5	.510E-02	8.1	.486E-02	6.9	.417E-02	5.3	.321E-02	5.9	.352E-02	3.5	.211E-02	2.4	.144E-02
90	2310	10.0	.543E-02	9.4	.511E-02	8.0	.434E-02	6.6	.358E-02	7.0	.378E-02	4.4	.239E-02	3.0	.160E-02
95	2390	10.2	.526E-02	9.6	.497E-02	8.2	.422E-02	6.8	.351E-02	7.2	.371E-02	4.6	.237E-02	3.1	.158E-02
100	2465	10.0	.500E-02	9.5	.473E-02	8.0	.400E-02	6.7	.332E-02	7.1	.353E-02	4.5	.224E-02	3.0	.150E-02
110	2600	9.2	.431E-02	8.7	.406E-02	7.2	.335E-02	5.9	.274E-02	6.2	.290E-02	3.8	.180E-02	2.6	.123E-02
120	2710	7.6	.336E-02	7.0	.313E-02	5.9	.263E-02	4.8	.214E-02	5.1	.225E-02	3.1	.137E-02	2.1	.938E-03
130	2795	6.1	.260E-02	5.7	.243E-02	4.9	.211E-02	4.0	.170E-02	4.2	.180E-02	2.5	.105E-02	1.7	.738E-03
140	2855	5.0	.209E-02	4.7	.197E-02	4.1	.171E-02	3.3	.139E-02	3.5	.146E-02	2.0	.854E-03	1.5	.605E-03
150	2900	4.4	.179E-02	4.1	.169E-02	3.5	.145E-02	2.9	.118E-02	3.1	.125E-02	1.8	.727E-03	1.3	.534E-03
160	2900	4.1	.167E-02	3.8	.156E-02	3.2	.131E-02	2.6	.108E-02	2.8	.114E-02	1.8	.734E-03	1.3	.531E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 6

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

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CORE
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A/A* =35.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: CN1 Qc	hc	Body Pt: CN2 Qc	hc	Body Pt: CN3 Qc	hc	Body Pt: CN4 Qc	hc	Body Pt: CB1 Qc	hc	Body Pt: CB2 Qc	hc	Body Pt: CB3 Qc	hc
80	2240	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
90	2310	2.5	.135E-02	2.0	.108E-02	1.5	.817E-03	1.5	.804E-03	1.9	.104E-02	1.5	.797E-03	0.9	.475E-03
95	2390	5.2	.271E-02	5.0	.259E-02	3.6	.185E-02	3.2	.164E-02	4.5	.234E-02	3.1	.160E-02	2.0	.102E-02
100	2465	10.1	.506E-02	9.0	.450E-02	5.2	.262E-02	4.8	.240E-02	7.1	.355E-02	4.5	.224E-02	3.0	.148E-02
105	2535	11.5	.555E-02	10.9	.526E-02	5.9	.284E-02	5.5	.266E-02	8.2	.395E-02	5.1	.248E-02	3.4	.166E-02
108	2575	11.6	.551E-02	11.0	.522E-02	5.9	.281E-02	5.6	.266E-02	8.3	.394E-02	5.2	.247E-02	3.5	.167E-02
110	2600	11.5	.539E-02	11.0	.511E-02	5.9	.274E-02	5.5	.257E-02	8.3	.386E-02	5.2	.241E-02	3.5	.164E-02
115	2655	11.1	.505E-02	10.5	.477E-02	5.6	.255E-02	5.2	.239E-02	8.0	.363E-02	4.9	.224E-02	3.4	.153E-02
120	2710	10.5	.466E-02	9.8	.437E-02	5.2	.233E-02	4.9	.217E-02	7.4	.327E-02	4.5	.202E-02	3.1	.136E-02
130	2795	8.7	.373E-02	8.0	.343E-02	4.4	.186E-02	4.0	.171E-02	6.3	.269E-02	3.7	.159E-02	2.5	.107E-02
140	2855	7.1	.296E-02	6.7	.278E-02	3.6	.151E-02	3.3	.139E-02	5.3	.223E-02	3.0	.127E-02	2.1	.892E-03
150	2900	6.0	.245E-02	5.7	.232E-02	3.1	.127E-02	2.9	.117E-02	4.6	.188E-02	2.6	.108E-02	1.9	.764E-03
160	2900	5.2	.213E-02	4.9	.202E-02	2.7	.112E-02	2.5	.104E-02	3.9	.161E-02	2.3	.958E-03	1.6	.667E-03
170	2900	4.8	.196E-02	4.5	.184E-02	2.5	.103E-02	2.3	.940E-03	3.5	.143E-02	2.2	.883E-03	1.5	.607E-03
180	2900	4.7	.191E-02	4.4	.180E-02	2.3	.958E-03	2.2	.885E-03	3.3	.134E-02	2.0	.823E-03	1.4	.563E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 7

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

++++++++++++++++++++++++
CORE
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A/A* =45.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

Alt ft 10**3	Tr R	Body Pt: CN1 Qc	hc	Body Pt: CN2 Qc	hc	Body Pt: CN3 Qc	hc	Body Pt: CN4 Qc	hc	Body Pt: CB1 Qc	hc	Body Pt: CB2 Qc	hc	Body Pt: CB3 Qc	hc
80	2145	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
85	2220	1.8	.106E-02	1.8	.104E-02	1.1	.639E-03	1.1	.625E-03	1.7	.971E-03	0.8	.474E-03	0.6	.345E-03
90	2310	5.2	.282E-02	5.0	.273E-02	2.6	.141E-02	2.5	.137E-02	3.9	.208E-02	2.1	.115E-02	1.4	.777E-03
95	2390	9.2	.478E-02	8.5	.440E-02	4.3	.225E-02	4.0	.209E-02	6.4	.333E-02	3.6	.188E-02	2.4	.125E-02
100	2465	10.6	.530E-02	10.0	.499E-02	5.3	.266E-02	5.0	.251E-02	7.5	.375E-02	4.7	.236E-02	3.1	.155E-02
103	2510	10.8	.527E-02	10.2	.495E-02	5.4	.267E-02	5.1	.250E-02	7.7	.371E-02	4.8	.235E-02	3.3	.159E-02
110	2600	10.4	.487E-02	10.0	.466E-02	5.1	.240E-02	4.8	.226E-02	7.4	.346E-02	4.5	.213E-02	3.1	.146E-02
120	2710	9.1	.405E-02	8.7	.388E-02	4.3	.189E-02	4.1	.181E-02	6.4	.286E-02	3.9	.171E-02	2.6	.116E-02
130	2795	7.7	.329E-02	7.3	.313E-02	3.4	.146E-02	3.3	.143E-02	5.2	.223E-02	3.0	.131E-02	2.0	.872E-03
140	2855	6.5	.272E-02	6.2	.260E-02	3.0	.124E-02	2.9	.121E-02	4.4	.185E-02	2.6	.109E-02	1.6	.682E-03
150	2900	5.6	.231E-02	5.4	.221E-02	2.7	.111E-02	2.6	.106E-02	3.9	.161E-02	2.3	.957E-03	1.4	.588E-03
160	2900	5.0	.204E-02	4.7	.194E-02	2.5	.103E-02	2.4	.969E-03	3.5	.144E-02	2.2	.896E-03	1.3	.548E-03
170	2900	4.5	.185E-02	4.3	.177E-02	2.3	.955E-03	2.2	.892E-03	3.3	.133E-02	2.1	.840E-03	1.3	.524E-03
180	2900	4.3	.178E-02	4.1	.168E-02	2.2	.915E-03	2.1	.847E-03	3.1	.125E-02	1.9	.790E-03	1.2	.499E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 8

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

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CORE
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A/A* =60.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

Alt ft 10**3	Tr R	Body Pt: CN1 Qc	hc	Body Pt: CN2 Qc	hc	Body Pt: CN3 Qc	hc	Body Pt: CN4 Qc	hc	Body Pt: CB1 Qc	hc	Body Pt: CB2 Qc	hc	Body Pt: CB3 Qc	hc
70	1930	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
80	2125	3.1	.186E-02	3.0	.181E-02	2.0	.121E-02	1.7	.102E-02	2.3	.140E-02	1.4	.864E-03	0.9	.522E-03
95	2220	5.7	.330E-02	5.1	.208E-02	2.8	.162E-02	3.7	.205E-02	3.8	.219E-02	2.5	.142E-02	1.5	.881E-03
90	2310	8.1	.437E-02	7.3	.395E-02	4.3	.233E-02	4.0	.217E-02	5.5	.298E-02	3.6	.196E-02	2.4	.131E-02
95	2390	9.7	.503E-02	9.0	.466E-02	4.8	.250E-02	4.8	.251E-02	6.8	.351E-02	4.3	.225E-02	2.8	.146E-02
100	2450	10.1	.504E-02	9.5	.474E-02	5.0	.250E-02	4.8	.238E-02	7.1	.355E-02	4.5	.226E-02	2.9	.147E-02
105	2535	9.9	.478E-02	9.4	.451E-02	4.9	.237E-02	4.6	.224E-02	6.9	.332E-02	4.4	.212E-02	2.9	.139E-02
110	2600	9.5	.445E-02	9.0	.419E-02	4.7	.219E-02	4.4	.207E-02	6.5	.304E-02	4.2	.196E-02	2.8	.130E-02
120	2710	8.4	.373E-02	7.8	.347E-02	4.0	.176E-02	3.7	.166E-02	5.6	.247E-02	3.5	.156E-02	2.4	.105E-02
130	2795	7.1	.303E-02	6.5	.280E-02	3.4	.145E-02	3.2	.137E-02	4.7	.199E-02	3.0	.128E-02	2.0	.838E-03
140	2850	5.8	.241E-02	5.5	.228E-02	3.0	.124E-02	2.8	.117E-02	3.9	.163E-02	2.6	.109E-02	1.6	.687E-03
150	2900	5.0	.204E-02	4.8	.196E-02	2.7	.110E-02	2.5	.102E-02	3.5	.142E-02	2.4	.974E-03	1.5	.596E-03
160	2900	4.5	.185E-02	4.3	.177E-02	2.4	.100E-02	2.2	.913E-03	3.2	.130E-02	2.1	.877E-03	1.3	.544E-03
170	2900	4.2	.171E-02	4.0	.165E-02	2.2	.903E-03	2.1	.841E-03	3.0	.124E-02	2.0	.810E-03	1.2	.493E-03
180	2900	4.0	.162E-02	3.8	.155E-02	2.0	.826E-03	1.9	.779E-03	2.8	.116E-02	1.8	.753E-03	1.2	.478E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

REMTECH

RL 91-61

ATTACHMENT 9.2

RTN 218-02

REMTECH TECHNICAL NOTE

TITLE: Addendum to RTN 218-01, "Preliminary Base Heating Environments for a Generalized ALS LO_2/LH_2 Launch Vehicle"

DATE: November 3, 1989

AUTHORS: Robert L. Bender and John E. Reardon

CONTRACT NO: NAS8-39141

PREPARED FOR: NASA/MSFC Induced Environment Branch ED-33

INTRODUCTION

Preliminary environments for multiple points located in the base of a generalized ALS LO_2/LH_2 launch vehicle were specified in REMTECH Technical Note RTN 218-01, published October 19, 1989. Subsequent discussions with MSFC ED-33 revealed a need to expand the environment determination to include additional locations on a "nacelle" type heat shield covering the engine power head as well as a planar heat shield attached directly to the thrust frame. This addendum includes these new environments in a format similar to the original publication.

ASSUMPTIONS AND GUIDELINES

The techniques and general methodology reported in RTN 218-01 were also utilized in this study. Engine arrangement and spacing were unchanged, and gimbaling was not considered. The environments were determined as a function of altitude with separate analyses required for radiation and plume induced convection.

CONFIGURATION AND BODY POINT SELECTION

The aft skirt and heat shield surrounding the engine nozzles which were considered in RTN 218-01 were removed for this later study. Individual nacelle heat shields covering each engine power head were added to both the booster and core elements as shown in Figs. 1 and 2. At the forward end of each nacelle, a heat shield was added extending laterally throughout the base at the gimbal plane.

Body points at critical locations on the aft face and sidewall of the nacelle were selected for the analysis. Additional body points on the base heat shield were also selected which correspond closely to the previous heat shield locations (reported in RTN 218-01) when the heat shield was further aft. These body points are shown on the booster and core schematics in Figs. 1 and 2, respectively.

RESULTS

As stated previously, radiation and convective base heating environments were determined separately and are presented in tabular form as a function of altitude. Separate environments were determined for the booster and core body points for each of the three LO_2/LH_2 nozzle area ratios of interest.

Predicted radiation rates for the Core and Booster vehicles are presented in Tables 1 and 2 for the base heat shield (CD-11/13 and BB-11/13) and the aft end of the nacelle (CP-11/14 and BP-11/16). Characteristics of the radiation will be discussed briefly in the following paragraphs.

The heat shield in this configuration is rather far forward of the nozzle exits, and it is shaded in some aspects by the nacelles. The peak point on the booster heat shield is higher than on the core vehicle because it can view more plumes, but as the radius of the heat shield location is increased, the booster rate drops relative to the core vehicle because views to the booster plumes become restricted by the relatively close spacing between outboard engines.

Radiation to the aft facing nacelle surfaces is slightly lower than previously reported heat shield rates at the same station because the points are shaded by being close to an engine nozzle. Radiation rates to the lateral facing surfaces of the nacelle are generally low because of the poor view of the plumes.

Cold wall convective heating rate and heat transfer coefficient plus base gas recovery temperature are presented in Tables 3 through 8. As expected, the aft corner surfaces of the nacelle facing inboard or toward an adjoining engine receive significant convective heating from reverse plume flow stagnation conditions. The base heat shield is sufficiently forward that the heating is attenuated; especially

since the removal of the aft skirt allows lateral relief from the reverse gases as they penetrate forward into the base region. General trends previously noted in RTN 218-01 are also evident in the new environments, i.e., core vehicle convective heating is less severe than the booster and has a shorter exposure time to the recirculated flow. Also, base interior heating is more severe than peripheral heating.

Table 1: ALS Booster Incident Plume Radiation Rates (BTU/sq-ft-sec)

BODY POINT	ALTITUDE (kFT) 0	10	30	50	100
35 NOZZLE AREA RATIO					
BB11	3.0	2.8	1.7	1.0	1.0
BB12	2.1	2.0	1.3	0.9	0.9
BB13	2.2	2.1	1.2	0.6	0.6
BP11	5.5	4.9	2.5	1.1	1.0
BP12	1.0	0.8	0.5	0.5	0.5
BP13	5.1	4.6	2.4	1.0	1.0
BP14	1.0	0.8	0.5	0.5	0.5
BP15	3.8	3.6	1.7	1.0	0.8
BP16	0.5	0.4	0.4	0.4	0.4
45 NOZZLE AREA RATIO					
BB11	2.7	2.5	1.5	0.9	0.9
BB12	1.8	1.8	1.2	0.8	0.8
BB13	2.0	1.8	1.1	0.6	0.6
BP11	5.0	4.5	2.3	1.0	0.9
BP12	0.8	0.7	0.4	0.4	0.4
BP13	4.7	4.2	2.2	0.9	0.9
BP14	0.8	0.7	0.4	0.4	0.4
BP15	3.5	3.3	1.6	0.9	0.8
BP16	0.4	0.4	0.3	0.3	0.3
60 NOZZLE AREA RATIO					
BB11	2.3	2.2	1.3	0.6	0.7
BB12	1.6	1.5	1.0	0.6	0.7
BB13	1.7	1.6	1.0	0.4	0.5
BP11	4.0	3.6	1.9	0.8	0.7
BP12	0.6	0.5	0.3	0.3	0.3
BP13	3.7	3.4	1.7	0.7	0.7
BP14	0.6	0.5	0.3	0.3	0.3
BP15	2.8	2.6	1.3	0.8	0.6
BP16	0.3	0.3	0.2	0.2	0.2

Table 2: ALS Core Incident Plume Radiation Rates (BTU/sq-ft-sec)

BODY POINT	ALTITUDE (kFT) 0	10	30	50	100
35 NOZZLE AREA RATIO					
CB11	2.4	2.7	2.4	1.9	1.2
CB12	2.8	2.9	2.4	1.7	0.9
CB13	3.6	3.6	2.7	1.7	0.5
CP11	3.6	3.6	2.7	1.7	0.9
CP12	1.0	0.8	0.5	0.5	0.5
CP13	3.7	3.8	3.2	2.2	0.9
CP14	0.7	0.7	0.5	0.4	0.4
45 NOZZLE AREA RATIO					
CB11	2.3	2.5	2.3	1.7	1.1
CB12	2.6	2.7	2.3	1.6	0.9
CB13	3.4	3.4	2.6	1.6	0.4
CP11	3.5	3.4	2.6	1.6	0.8
CP12	0.9	0.7	0.5	0.5	0.5
CP13	3.5	3.7	3.1	2.1	0.8
CP14	0.6	0.6	0.5	0.3	0.3
60 NOZZLE AREA RATIO					
CB11	2.1	2.3	2.1	1.6	1.0
CB12	2.4	2.5	2.1	1.4	0.8
CB13	3.1	3.1	2.3	1.4	0.4
CP11	3.3	3.3	2.5	1.5	0.8
CP12	0.7	0.6	0.4	0.4	0.4
CP13	3.4	3.5	2.9	2.0	0.8
CP14	0.6	0.5	0.4	0.3	0.3

Table 3:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

BOOSTER

A/A* =35.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: BP11 Qc	hc	Body Pt: BP12 Qc	hc	Body Pt: BP13 Qc	hc	Body Pt: BP14 Qc	hc	Body Pt: BP15 Qc	hc	Body Pt: BP16 Qc	hc
58	1605	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
60	1660	0.6	.523E-03	0.5	.416E-03	0.4	.343E-03	0.3	.248E-03	0.2	.167E-03	0.1	.833E-04
70	1910	3.4	.235E-02	2.7	.183E-02	2.3	.161E-02	1.5	.107E-02	1.5	.104E-02	1.1	.743E-03
80	2125	6.8	.410E-02	5.6	.336E-02	5.1	.309E-02	4.4	.267E-02	3.0	.178E-02	2.9	.177E-02
90	2310	9.7	.523E-02	8.8	.475E-02	8.3	.451E-02	7.3	.394E-02	6.1	.330E-02	5.1	.278E-02
100	2465	10.2	.509E-02	9.4	.467E-02	8.9	.446E-02	8.1	.402E-02	6.7	.334E-02	5.7	.283E-02
110	2600	8.9	.416E-02	8.0	.373E-02	7.3	.340E-02	6.3	.297E-02	5.6	.262E-02	4.7	.219E-02
120	2710	6.6	.292E-02	6.0	.266E-02	5.5	.245E-02	4.9	.217E-02	4.2	.187E-02	3.5	.155E-02
130	2795	5.3	.227E-02	4.9	.212E-02	4.6	.197E-02	4.1	.176E-02	3.5	.150E-02	2.9	.124E-02
140	2855	4.6	.192E-02	4.3	.180E-02	4.1	.170E-02	3.6	.151E-02	3.1	.129E-02	2.6	.108E-02
150	2900	4.1	.170E-02	3.8	.157E-02	3.7	.150E-02	3.3	.135E-02	2.8	.113E-02	2.3	.942E-03

ALT ft 10**3	Tr R	Body Pt: BB11 Qc	hc	Body Pt: BB12 Qc	hc	Body Pt: BB13 Qc	hc
58	1605	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
60	1660	0.1	.833E-04	0.0	.000E+00	0.0	.000E+00
70	1910	0.7	.460E-03	0.3	.195E-03	0.2	.150E-03
80	2125	1.6	.949E-03	0.8	.464E-03	0.6	.341E-03
90	2310	2.5	.134E-02	1.4	.732E-03	1.0	.545E-03
100	2465	2.6	.128E-02	1.6	.786E-03	1.1	.543E-03
110	2600	2.2	.103E-02	1.3	.612E-03	0.9	.408E-03
120	2710	1.8	.789E-03	1.0	.453E-03	0.7	.305E-03
130	2795	1.4	.614E-03	0.8	.345E-03	0.6	.246E-03
140	2855	1.2	.519E-03	0.7	.279E-03	0.5	.209E-03
150	2900	1.1	.448E-03	0.6	.243E-03	0.4	.161E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 4:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

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BOOSTER
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A/A* =45.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: BP11 Qc	hc	Body Pt: BP12 Qc	hc	Body Pt: BP13 Qc	hc	Body Pt: BP14 Qc	hc	Body Pt: BP15 Qc	hc	Body Pt: BP16 Qc	hc
45	1240	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
50	1385	0.6	.697E-03	0.5	.545E-03	0.4	.446E-03	0.3	.309E-03	0.2	.185E-03	0.2	.216E-03
60	1660	2.3	.189E-02	1.9	.158E-02	1.8	.148E-02	1.4	.113E-02	0.9	.771E-03	0.6	.526E-03
70	1910	4.1	.285E-02	3.6	.245E-02	3.3	.231E-02	2.8	.195E-02	2.1	.143E-02	1.6	.111E-02
80	2125	6.3	.378E-02	5.4	.323E-02	5.1	.309E-02	4.5	.272E-02	3.5	.212E-02	2.8	.170E-02
90	2310	7.7	.417E-02	7.0	.379E-02	6.7	.364E-02	6.2	.334E-02	5.0	.269E-02	4.2	.224E-02
95	2390	7.9	.409E-02	7.3	.378E-02	7.1	.368E-02	6.4	.332E-02	5.3	.275E-02	4.4	.228E-02
100	2465	7.8	.390E-02	7.2	.357E-02	6.8	.341E-02	6.2	.311E-02	5.1	.253E-02	4.3	.216E-02
110	2600	7.1	.332E-02	6.2	.288E-02	5.9	.275E-02	5.3	.247E-02	4.3	.201E-02	3.6	.168E-02
120	2710	5.7	.256E-02	5.0	.222E-02	4.8	.212E-02	4.2	.185E-02	3.4	.151E-02	2.8	.124E-02
130	2795	4.7	.199E-02	4.2	.179E-02	4.0	.170E-02	3.5	.149E-02	2.8	.121E-02	2.3	.979E-03
140	2855	3.9	.162E-02	3.5	.147E-02	3.4	.140E-02	3.0	.126E-02	2.5	.104E-02	2.0	.825E-03
150	2900	3.3	.137E-02	3.1	.128E-02	3.0	.123E-02	2.7	.112E-02	2.3	.930E-03	1.8	.741E-03
160	2900	3.1	.127E-02	2.9	.119E-02	2.8	.116E-02	2.5	.104E-02	2.1	.866E-03	1.8	.719E-03

ALT ft 10**3	Tr R	Body Pt: BB11 Qc	hc	Body Pt: BB12 Qc	hc	Body Pt: BB13 Qc	hc
45	1240	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
50	1385	0.1	.108E-03	0.0	.000E+00	0.0	.000E+00
60	1660	0.3	.259E-03	0.2	.130E-03	0.1	.877E-04
70	1910	0.7	.496E-03	0.4	.284E-03	0.3	.195E-03
80	2125	1.2	.739E-03	0.7	.431E-03	0.5	.292E-03
90	2310	1.8	.996E-03	1.1	.595E-03	0.7	.401E-03
95	2390	2.0	.104E-02	1.3	.674E-03	0.9	.440E-03
100	2465	1.9	.948E-03	1.2	.605E-03	0.8	.407E-03
110	2600	1.7	.812E-03	1.0	.483E-03	0.7	.315E-03
120	2710	1.4	.635E-03	0.8	.373E-03	0.6	.280E-03
130	2795	1.2	.507E-03	0.7	.303E-03	0.5	.194E-03
140	2855	1.0	.429E-03	0.6	.252E-03	0.4	.172E-03
150	2900	0.9	.363E-03	0.5	.219E-03	0.3	.141E-03
160	2900	0.8	.325E-03	0.4	.169E-03	0.2	.101E-03

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 5:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

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BOOSTER
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A/A* =60.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: BP11 Qc	hc	Body Pt: BP12 Qc	hc	Body Pt: BP13 Qc	hc	Body Pt: BP14 Qc	hc	Body Pt: BP15 Qc	hc	Body Pt: BP16 Qc	hc
40	1095	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
45	1240	0.5	.628E-03	0.4	.500E-03	0.3	.372E-03	0.2	.308E-03	0.2	.295E-03	0.1	.128E-03
50	1385	0.9	.949E-03	0.7	.741E-03	0.6	.639E-03	0.5	.519E-03	0.4	.449E-03	0.3	.352E-03
60	1660	2.0	.165E-02	1.7	.142E-02	1.5	.128E-02	1.4	.114E-02	1.1	.947E-03	0.9	.765E-03
70	1910	3.3	.229E-02	3.0	.205E-02	2.8	.191E-02	2.5	.171E-02	2.0	.139E-02	1.7	.115E-02
80	2125	5.0	.297E-02	4.5	.267E-02	4.1	.249E-02	3.8	.226E-02	3.0	.179E-02	2.6	.155E-02
90	2310	6.1	.332E-02	5.6	.305E-02	5.4	.289E-02	4.9	.263E-02	4.0	.216E-02	3.5	.187E-02
95	2390	6.2	.321E-02	5.7	.295E-02	5.5	.284E-02	5.0	.259E-02	4.1	.212E-02	3.6	.187E-02
100	2465	5.9	.295E-02	5.5	.275E-02	5.2	.261E-02	4.7	.236E-02	3.9	.196E-02	3.2	.162E-02
110	2600	5.2	.245E-02	4.8	.226E-02	4.5	.211E-02	4.0	.189E-02	3.4	.157E-02	2.7	.125E-02
120	2710	4.4	.197E-02	4.1	.184E-02	3.8	.168E-02	3.4	.149E-02	2.7	.122E-02	2.2	.987E-03
130	2795	3.8	.162E-02	3.5	.150E-02	3.2	.139E-02	2.9	.122E-02	2.3	.990E-03	1.9	.808E-03
140	2855	3.2	.135E-02	3.0	.125E-02	2.8	.117E-02	2.5	.104E-02	2.0	.826E-03	1.7	.692E-03
150	2900	2.8	.115E-02	2.6	.106E-02	2.4	.990E-03	2.2	.885E-03	1.7	.711E-03	1.5	.606E-03
160	2900	2.5	.101E-02	2.2	.922E-03	2.1	.859E-03	1.9	.796E-03	1.5	.632E-03	1.3	.548E-03

ALT ft 10**3	Tr R	Body Pt: BB11 Qc	hc	Body Pt: BB12 Qc	hc	Body Pt: BB13 Qc	hc
40	1095	0.0	.000E+00	0.0	.000E+00	0.0	.000E+00
45	1240	0.1	.128E-03	0.0	.000E+00	0.0	.000E+00
50	1385	0.2	.238E-03	0.0	.000E+00	0.0	.000E+00
60	1660	0.4	.358E-03	0.2	.187E-03	0.1	.101E-03
70	1910	0.8	.536E-03	0.5	.315E-03	0.3	.182E-03
80	2125	1.2	.714E-03	0.7	.413E-03	0.4	.259E-03
90	2310	1.5	.789E-03	0.9	.491E-03	0.6	.317E-03
95	2390	1.6	.829E-03	1.0	.518E-03	0.6	.337E-03
100	2465	1.4	.716E-03	0.9	.440E-03	0.6	.299E-03
110	2600	1.2	.551E-03	0.7	.335E-03	0.5	.227E-03
120	2710	1.0	.443E-03	0.6	.273E-03	0.4	.174E-03
130	2795	0.9	.385E-03	0.5	.214E-03	0.3	.137E-03
140	2855	0.8	.326E-03	0.4	.188E-03	0.3	.108E-03
150	2900	0.7	.282E-03	0.3	.131E-03	0.2	.864E-04
160	2900	0.6	.232E-03	0.3	.127E-03	0.2	.810E-04

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 6:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

CORE

A/A* =35.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: CP11 Qc	hc	Body Pt: CP12 Qc	hc	Body Pt: CP13 Qc	hc	Body Pt: CP14 Qc	hc
86	2240	0.0	.000E+00	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00
90	2310	1.4	.733E-03	0.8	.435E-03	0.5	.275E-03	0.3	.158E-03
95	2390	3.4	.176E-02	2.5	.130E-02	1.6	.829E-03	1.1	.570E-03
100	2465	5.5	.276E-02	4.5	.223E-02	2.9	.144E-02	2.1	.103E-02
105	2535	6.3	.304E-02	5.7	.275E-02	4.1	.198E-02	3.3	.159E-02
108	2575	6.4	.303E-02	5.9	.279E-02	4.3	.203E-02	3.7	.173E-02
110	2600	6.3	.296E-02	5.8	.271E-02	4.2	.196E-02	3.6	.168E-02
115	2655	6.1	.278E-02	5.5	.251E-02	4.1	.187E-02	3.5	.152E-02
120	2710	5.5	.246E-02	4.9	.219E-02	3.6	.160E-02	3.2	.142E-02
130	2795	4.6	.197E-02	4.2	.179E-02	3.1	.134E-02	2.6	.113E-02
140	2855	3.9	.165E-02	3.6	.152E-02	2.7	.113E-02	2.3	.947E-03
150	2900	3.5	.142E-02	3.2	.130E-02	2.4	.980E-03	2.0	.802E-03
160	2900	3.1	.127E-02	2.8	.115E-02	2.1	.874E-03	1.7	.711E-03
170	2900	2.8	.114E-02	2.5	.103E-02	1.9	.782E-03	1.5	.630E-03
180	2900	2.6	.105E-02	2.3	.933E-03	1.7	.712E-03	1.4	.571E-03

ALT ft 10**3	Tr R	Body Pt: CB11 Qc	hc	Body Pt: CB12 Qc	hc	Body Pt: CB13 Qc	hc
86	2240	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00
90	2310	0.2	.811E-04	0.1	.541E-04	0.1	.541E-04
95	2390	0.5	.259E-03	0.3	.154E-03	0.2	.104E-03
100	2465	1.0	.508E-03	0.7	.329E-03	0.4	.188E-03
105	2535	1.5	.747E-03	1.0	.482E-03	0.6	.265E-03
108	2575	1.7	.804E-03	1.2	.567E-03	0.7	.331E-03
110	2600	1.6	.731E-03	1.1	.498E-03	0.6	.254E-03
115	2655	1.5	.683E-03	1.0	.456E-03	0.6	.273E-03
120	2710	1.3	.591E-03	0.9	.386E-03	0.5	.216E-03
130	2795	1.1	.480E-03	0.7	.315E-03	0.4	.162E-03
140	2855	1.0	.412E-03	0.6	.252E-03	0.3	.129E-03
150	2900	0.8	.340E-03	0.5	.224E-03	0.3	.109E-03
160	2900	0.7	.306E-03	0.5	.196E-03	0.2	.913E-04
170	2900	0.7	.273E-03	0.4	.152E-03	0.2	.895E-04
180	2900	0.5	.256E-03	0.3	.140E-03	0.2	.772E-04

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 7:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

CORE

A/A* =45.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: CP11 Qc	hc	Body Pt: CP12 Qc	hc	Body Pt: CP13 Qc	hc	Body Pt: CP14 Qc	hc
80	2145	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00
85	2220	1.2	.682E-03	1.0	.568E-03	0.7	.398E-03	0.5	.284E-03
90	2310	3.0	.161E-02	2.4	.128E-02	1.7	.906E-03	1.3	.704E-03
95	2390	4.8	.249E-02	4.1	.212E-02	2.9	.150E-02	2.2	.117E-02
100	2465	5.8	.287E-02	5.3	.264E-02	3.7	.187E-02	3.2	.158E-02
103	2510	5.9	.288E-02	5.4	.263E-02	4.0	.195E-02	3.3	.161E-02
110	2600	5.7	.267E-02	5.2	.244E-02	3.8	.178E-02	3.2	.149E-02
120	2710	5.1	.226E-02	4.6	.203E-02	3.3	.145E-02	2.7	.121E-02
130	2795	4.3	.182E-02	3.9	.165E-02	2.8	.118E-02	2.3	.997E-03
140	2855	3.6	.152E-02	3.4	.141E-02	2.4	.101E-02	2.0	.843E-03
150	2900	3.2	.132E-02	3.0	.122E-02	2.2	.890E-03	1.8	.727E-03
160	2900	2.9	.119E-02	2.7	.110E-02	2.0	.800E-03	1.6	.658E-03
170	2900	2.6	.108E-02	2.4	.100E-02	1.8	.726E-03	1.5	.605E-03
180	2900	2.4	.100E-02	2.3	.926E-03	1.7	.678E-03	1.4	.568E-03

ALT ft 10**3	Tr R	Body Pt: CB11 Qc	hc	Body Pt: CB12 Qc	hc	Body Pt: CB13 Qc	hc
80	2145	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00
85	2220	0.2	.114E-03	0.1	.568E-04	0.0	.000E-00
90	2310	0.7	.371E-03	0.4	.218E-03	0.2	.135E-03
95	2390	1.1	.596E-03	0.7	.363E-03	0.5	.259E-03
100	2465	1.6	.778E-03	1.0	.483E-03	0.6	.323E-03
103	2510	1.7	.829E-03	1.2	.585E-03	0.7	.337E-03
110	2600	1.5	.722E-03	1.0	.446E-03	0.6	.302E-03
120	2710	1.3	.567E-03	0.8	.377E-03	0.5	.218E-03
130	2795	1.0	.436E-03	0.7	.319E-03	0.4	.161E-03
140	2855	0.9	.386E-03	0.6	.247E-03	0.3	.124E-03
150	2900	0.8	.347E-03	0.5	.200E-03	0.3	.106E-03
160	2900	0.7	.274E-03	0.4	.179E-03	0.2	.895E-04
170	2900	0.6	.247E-03	0.4	.168E-03	0.2	.840E-04
180	2900	0.6	.257E-03	0.4	.173E-03	0.2	.731E-04

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

Table 8:

ADVANCED LAUNCH SYSTEM
COLD WALL CONVECTIVE BASE HEATING

CORE

A/A* =60.0

LO2-LH2 Engines - Pchamber= 2250 PSIA

ALT ft 10**3	Tr R	Body Pt: CP11 Qc	Body Pt: CP11 hc	Body Pt: CP12 Qc	Body Pt: CP12 hc	Body Pt: CP13 Qc	Body Pt: CP13 hc	Body Pt: CP14 Qc	Body Pt: CP14 hc
70	0	0.0	.000E+00	0.0	.000E-00	0.0	.000E-00	0.0	.000E-00
80	2125	1.5	.903E-03	1.2	.711E-03	0.9	.512E-03	0.6	.389E-03
85	2220	2.6	.148E-02	2.2	.125E-02	1.6	.909E-03	1.2	.710E-03
90	2310	4.1	.221E-02	3.3	.179E-02	2.4	.132E-02	2.0	.108E-02
95	2390	5.2	.272E-02	4.6	.238E-02	3.3	.171E-02	2.8	.142E-02
100	2465	5.5	.275E-02	5.1	.254E-02	3.7	.184E-02	3.2	.160E-02
105	2535	5.4	.260E-02	5.0	.241E-02	3.6	.173E-02	3.1	.149E-02
110	2600	5.1	.239E-02	4.7	.217E-02	3.4	.159E-02	2.9	.138E-02
120	2710	4.4	.195E-02	4.0	.177E-02	2.9	.128E-02	2.4	.109E-02
130	2795	3.8	.162E-02	3.4	.148E-02	2.5	.105E-02	2.1	.891E-03
140	2855	3.3	.138E-02	3.0	.127E-02	2.1	.896E-03	1.8	.747E-03
150	2900	3.0	.121E-02	2.7	.112E-02	1.9	.787E-03	1.6	.640E-03
160	2900	2.7	.109E-02	2.4	.997E-03	1.7	.709E-03	1.4	.578E-03
170	2900	2.4	.977E-03	2.2	.914E-03	1.6	.653E-03	1.3	.527E-03
180	2900	2.2	.889E-03	2.0	.837E-03	1.5	.607E-03	1.2	.502E-03

ALT ft 10**3	Tr R	Body Pt: CB11 Qc	Body Pt: CB11 hc	Body Pt: CB12 Qc	Body Pt: CB12 hc	Body Pt: CB13 Qc	Body Pt: CB13 hc
70	0	0.0	.000E+00	0.0	.000E+00	0.0	.000E-00
80	2125	0.3	.182E-03	0.2	.121E-03	0.1	.823E-04
85	2220	0.6	.341E-03	0.4	.227E-03	0.2	.114E-03
90	2310	1.0	.545E-03	0.6	.331E-03	0.4	.207E-03
95	2390	1.3	.674E-03	0.8	.415E-03	0.5	.259E-03
100	2465	1.5	.748E-03	1.0	.494E-03	0.6	.299E-03
105	2535	1.4	.675E-03	0.9	.434E-03	0.5	.241E-03
110	2600	1.3	.617E-03	0.9	.408E-03	0.5	.229E-03
120	2710	1.1	.475E-03	0.8	.338E-03	0.4	.174E-03
130	2795	1.0	.419E-03	0.7	.283E-03	0.4	.158E-03
140	2855	0.9	.373E-03	0.6	.251E-03	0.3	.117E-03
150	2900	0.8	.325E-03	0.5	.221E-03	0.3	.111E-03
160	2900	0.6	.259E-03	0.5	.196E-03	0.2	.966E-04
170	2900	0.5	.239E-03	0.4	.166E-03	0.2	.820E-04
180	2900	0.5	.235E-03	0.4	.156E-03	0.2	.884E-04

units for Qc: BTU/ft2-sec
units for hc: BTU/ft2-sec-R

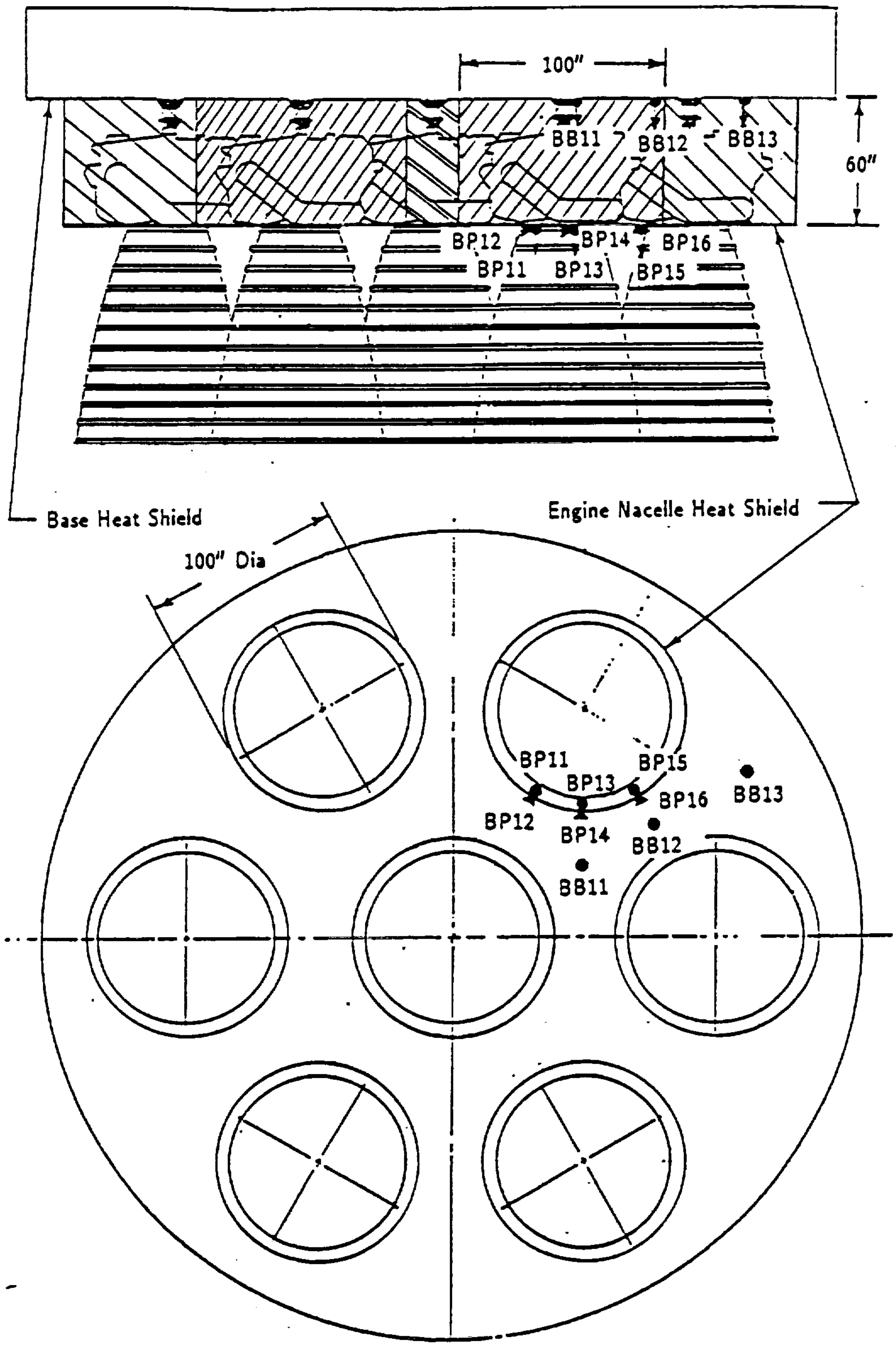

Figure 1: Booster Locations Selected for Base Heating Analysis

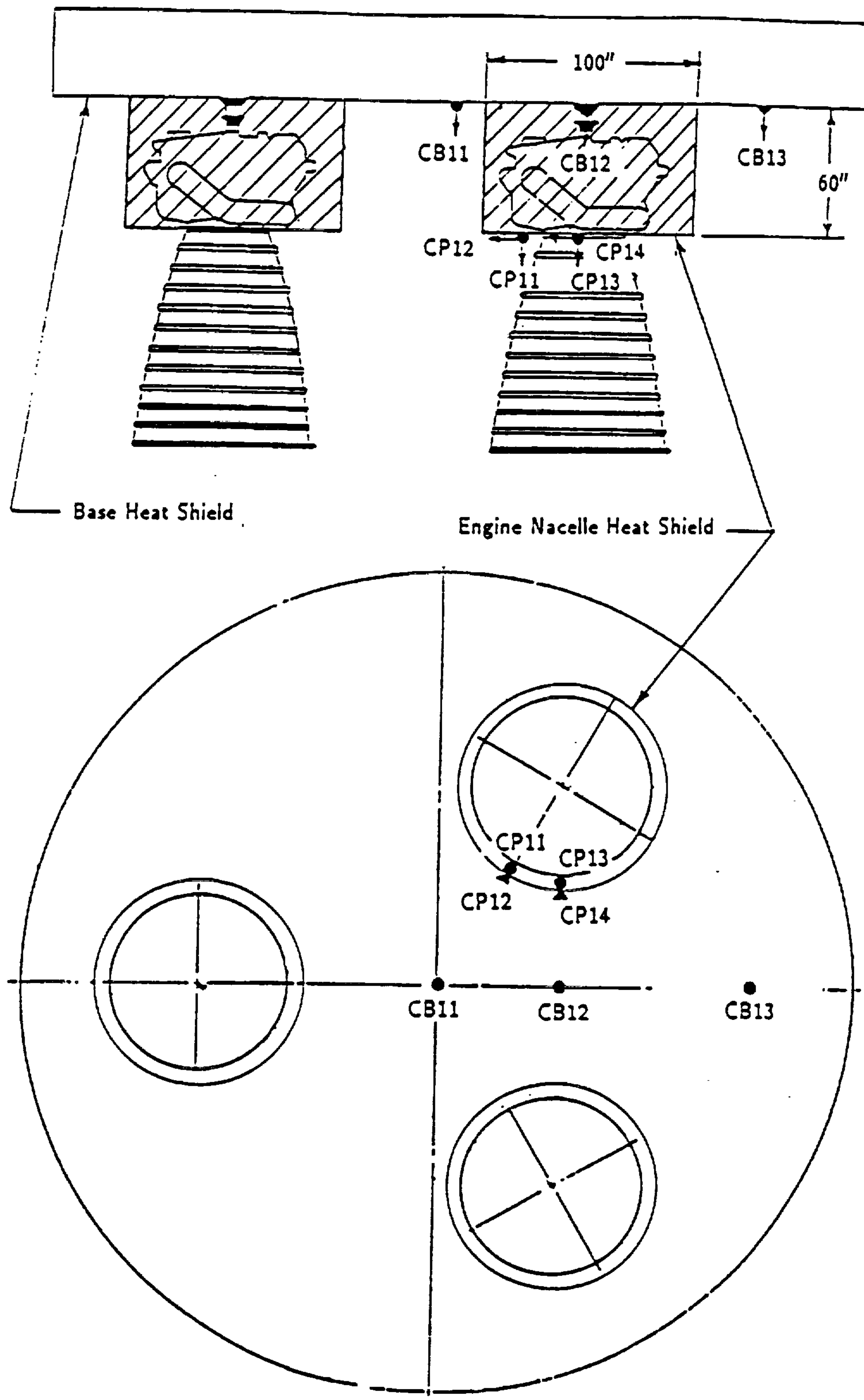

Figure 2: Core Locations Selected for Base Heating Analysis

Table 1: ALS Booster Incident Plume Radiation Rates (Btu/sq-ft-sec)

ALS BOOSTER

INCIDENT PLUME RADIATION RATES (Btu/sq-ft-sec)

35 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT)				
	0	10	30	50	100
BB1	7.0	6.6	4.8	2.5	1.3
BB2	7.0	6.6	4.8	2.5	1.3
BB3	7.9	7.4	4.5	2.3	0.7
BN1	21.0	18.0	11.7	9.2	7.0
BN2	10.5	9.0	4.5	3.5	3.0
BS1	13.4	12.	6.2	4.2	1.6
BS2	3.7	3.3	1.5	0.8	0.6
45 NOZZLE AREA RATIO					
BODY POINT	ALTITUDE (kFT)				
	0	10	30	50	100
BB1	6.4	6.1	4.1	2.2	1.3
BB2	6.8	6.4	3.7	1.9	1.1
BB3	6.8	6.4	3.7	1.9	1.1
BN1	28.0	24.0	12.0	8.5	6.0
BN2	14.0	12.5	4.9	3.5	3.0
BS1	13.4	12.0	6.2	4.2	1.6
BS2	3.7	3.3	1.5	0.8	0.6

REMTECH

NLS
PRELIMINARY CYCLE 1
ASCENT BASE HEATING
ENVIRONMENTS

SEPTEMBER 26, 1991

NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

PREPARED BY:
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NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

PROBLEM DEFINITION

Ascent base heating is a plume induced environment occurring throughout powered ascent. It is a combination of plume radiation and convection occurring when plume gases are recirculated into the base. Both heating modes are basically a function of altitude, with flight-time effects also entering through variations in engine thrust.

•*HLLV BASE REGION*

Plume induced base heating environments will be generally applicable to all aft surface of the ASRB forward to the attach ring. Base heating to the core vehicle will affect all base region surfaces forward to the heat shield from lift-off until plume induced separation (PIFS) occurs. After PIFS, reversed flow convection and local gas radiation will occur within the separated region, which may extend one-half of the core tank length forward of the heat shield.

•*1.5 STAGE BASE REGION*

The general base region including the STME nozzles and associated hardware will receive plume radiation and reversed gas convection from lift-off to PIFS. After PIFS initiates (which may be earlier in the 1.5 stage because of the two additional engines and larger composite plume), convection and local radiation will occur within the separated region. PIFS will be drastically altered or eliminated when the outboard engines are jettisoned and only the core sustainer engines are firing. Heating during this period will be minimal and confined to the area around the sustainer engine nozzles.

TECHNICAL APPROACH

PRELIMINARY CYCLE 1 ENVIRONMENTS

- Two (2) body points per vehicle
- May, 1991 engine out trajectories
- May 28, 1991 Reference configuration and performance data
- Simplified methodology (conservative)
- Did not consider STME turbine exhaust discharge
- Did not consider plume induced flow separation
- Output by September 17, 1991

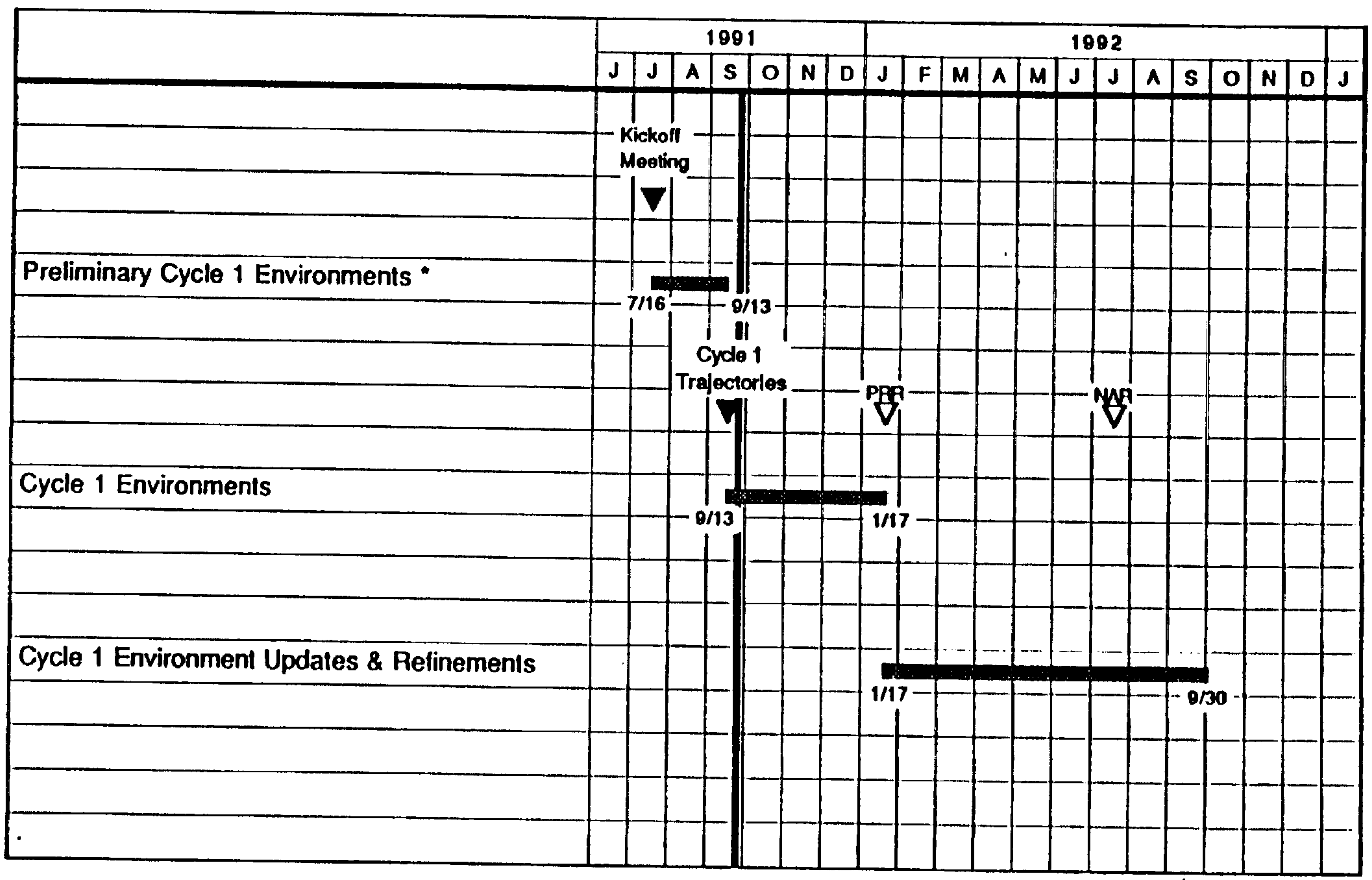
CYCLE 1 ENVIRONMENTS

- 12 to 15 body points per vehicle
- Tailored trajectories for maximum plume heating
- Fall 1991 reference configuration and performance data
- Improved methodology utilizing updated plume definitions
- Quantify effects of STME turbine exhaust disposal schemes
- Quantify effect of plume induced flow separation
- Output by January 13, 1992

NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

TASK 3 - FM - 006

NLS BASE HEATING ANALYSIS



* Reported in REMTECH RTN 218-03, dated Sept. 13, 1991.

UNCERTAINTIES IN THE OUTPUT

•*BACKGROUND*

Normally, plume predictions which define the flowfields and properties necessary to support impingement and ascent base heating studies are quite accurate — although difficult to produce for long distances downstream. Multiple plume interaction regions and 3–D base flowfields for higher altitudes are not attempted analytically due to their complexity and because they change constantly throughout ascent. Base heating prediction accuracy varies depending on the base geometry, number of plumes, type of plumes, etc. for radiation; and upon the extent and applicability of the flight and model database for convection. Approximately 20% uncertainty is customary in most base heating environment predictions.

•*NLS PLUME DEFINITION AND BASE HEATING*

The single biggest uncertainty factor in the NLS plume prediction and base heating analysis is the lack of definitive data describing the turbine exhaust disposal scheme in the STME nozzle. Large variations in accuracy of the plume viscous mixing layer composition, thermodynamic and transport properties could occur depending on the flow rates of injectants or by-pass flows. These variations influence our ability to characterize plume radiation models and heating potential of reversed flows into the base region — which directly affect the magnitudes of the base heating environments.

•It is assumed that the base geometry and propulsion/performance parameters will not vary significantly during this study, so the environment should be accurate within the methodology uncertainty (i.e.≈ 20% plus the turbine exhaust uncertainty).

PRELIMINARY CYCLE 1 METHODOLOGY

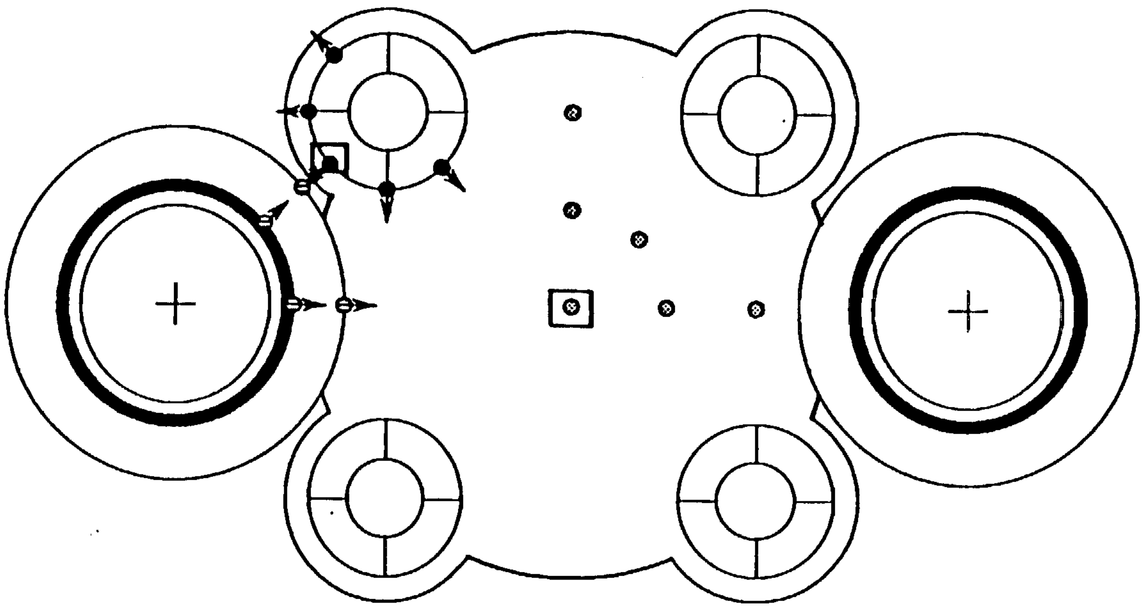
$$Q_{Total} = Q_{Rad} + Q_{Conv}$$

- ***RADIATION***
  - ASRM:
    - Viewfactor predictions using Cycle 1 sea-level plume model
    - Modified Cycle 1 altitude adjustment function
    - Modified Cycle 1 shutdown spike adjustment function
  - STME:
    - Band-model predictions on scaled plumes (0–160 kft).
    - Estimated afterburning increase
    - Estimated base burning radiation
    - Estimated plume interference effects
- ***CONVECTION***
  - PLUME INTERACTIONS: From preliminary plume studies
  - INCIPIENT RECIRCULATION: Based upon engine spacing empirical study
  - CHOKED FLOW ALTITUDE: Empirical, TND-1093
  - STME RECIRCULATION: From scaled data base (Shuttle Orbiter, Saturn V S-11 Stage S-I S-IV Stage)
  - ASRB RECIRCULATION: From Shuttle data base and ASRB Cycle 1 methodology

NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

BODY POINT LOCATIONS

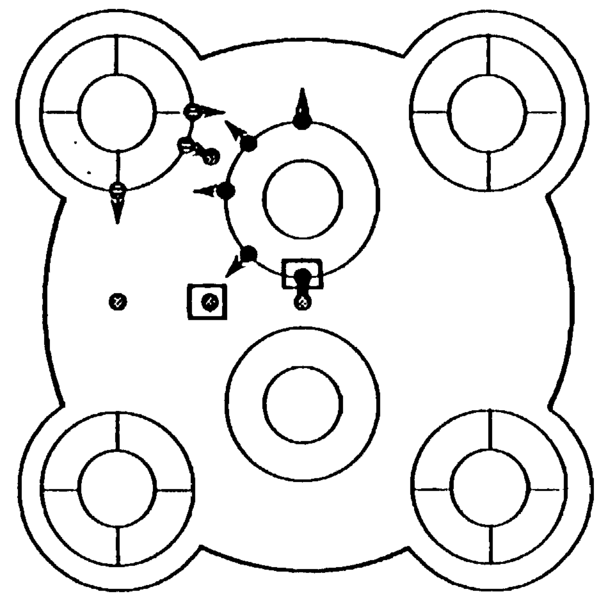
IN-LINE HLLV



SUMMARY

- ⊖ - ASRB (4)
- ● - OUTBOARD STME (5)
- ⊗ - CORE HEAT SHIELD (6)

1.5 STAGE REFERENCE



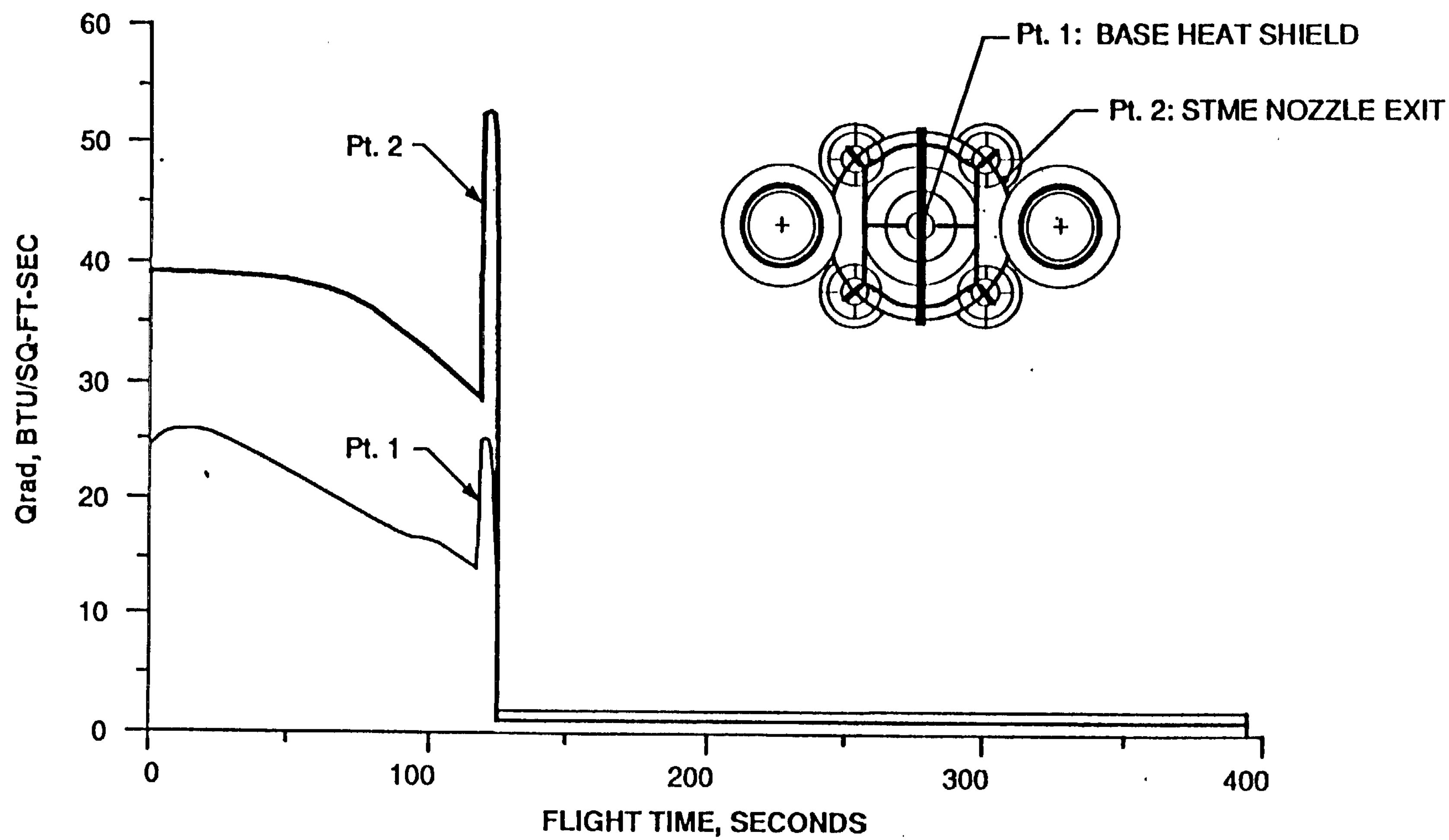
SUMMARY

- ⊖ - OUTBOARD STME (3)
- ● - INBOARD STME (5)
- ⊗ - CORE HEAT SHIELD (4)

NOTE: BOXED POINTS INVESTIGATED IN PRELIMINARY CYCLE 1 ANALYSIS

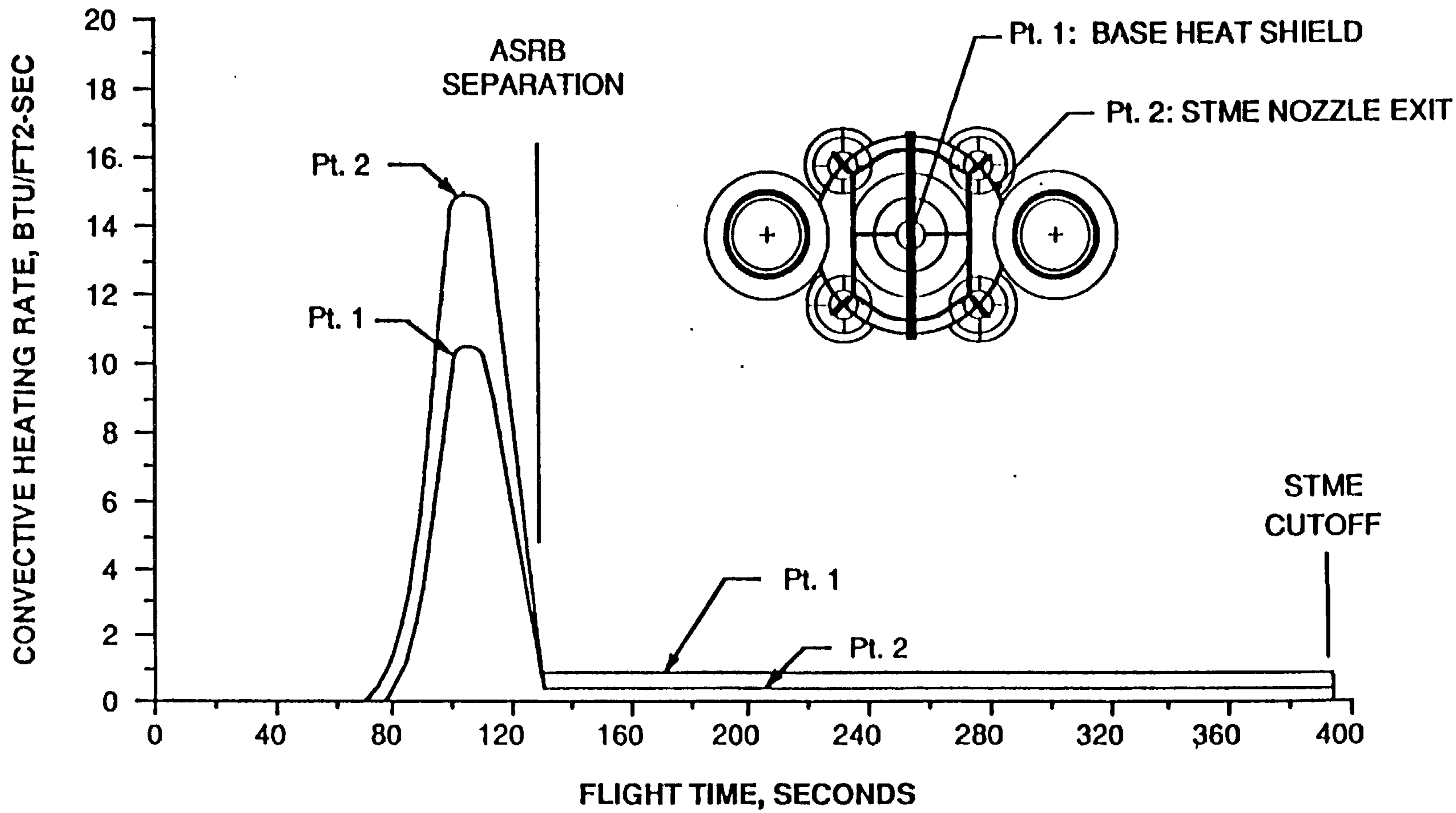
NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

IN-LINE HLLV REFERENCE



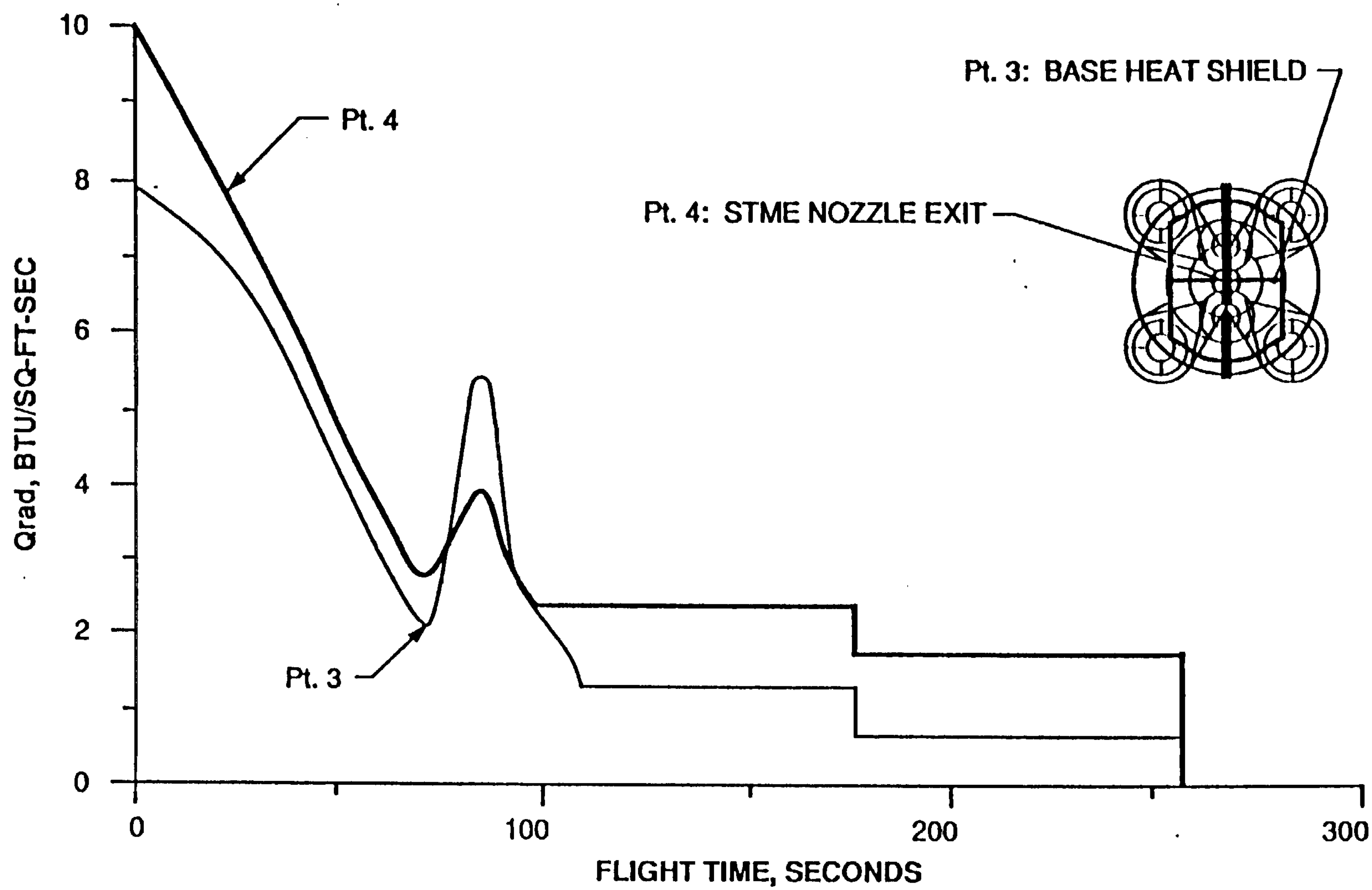
NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

IN-LINE HLLV REFERENCE



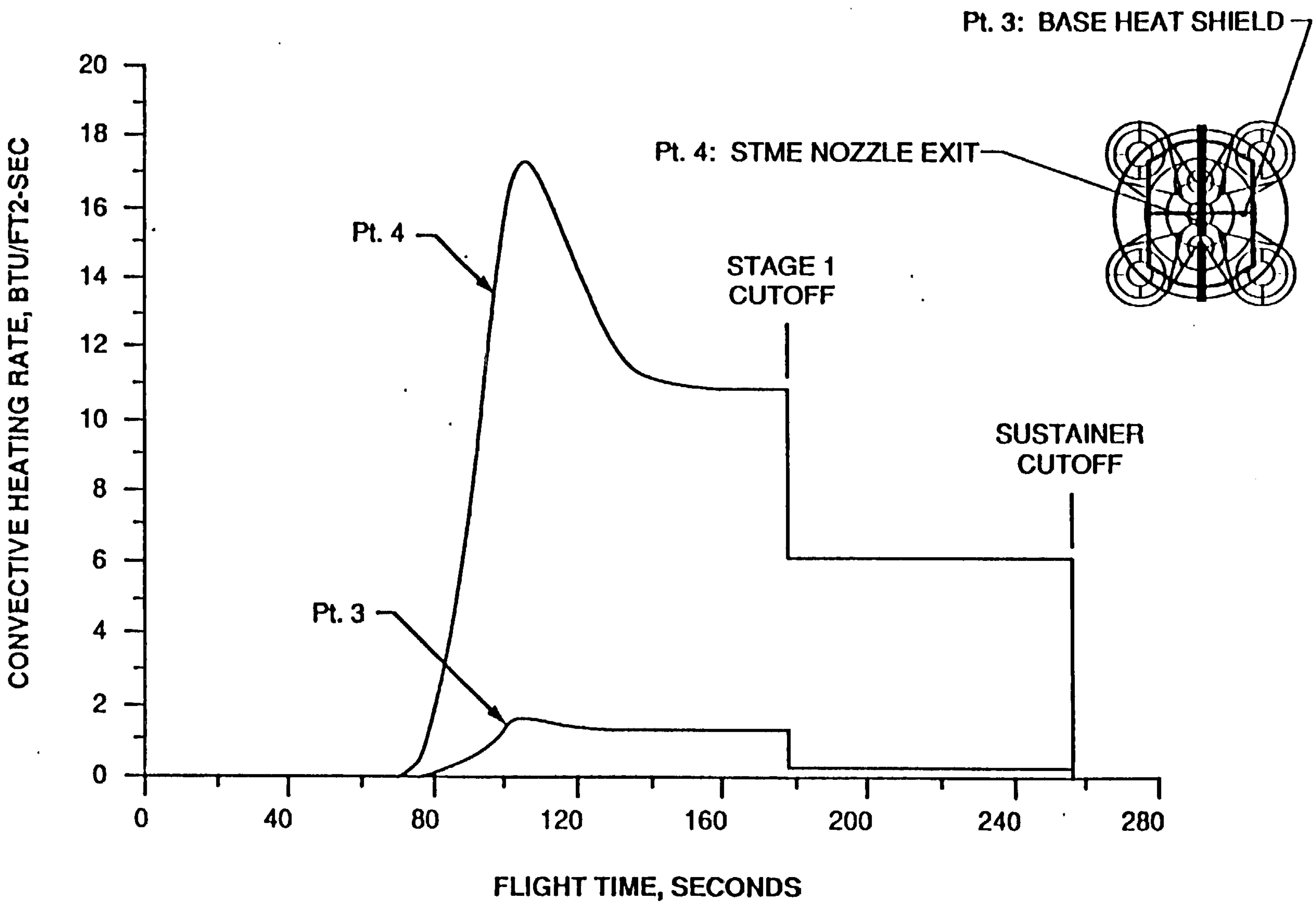
NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

1.5 STAGE REFERENCE



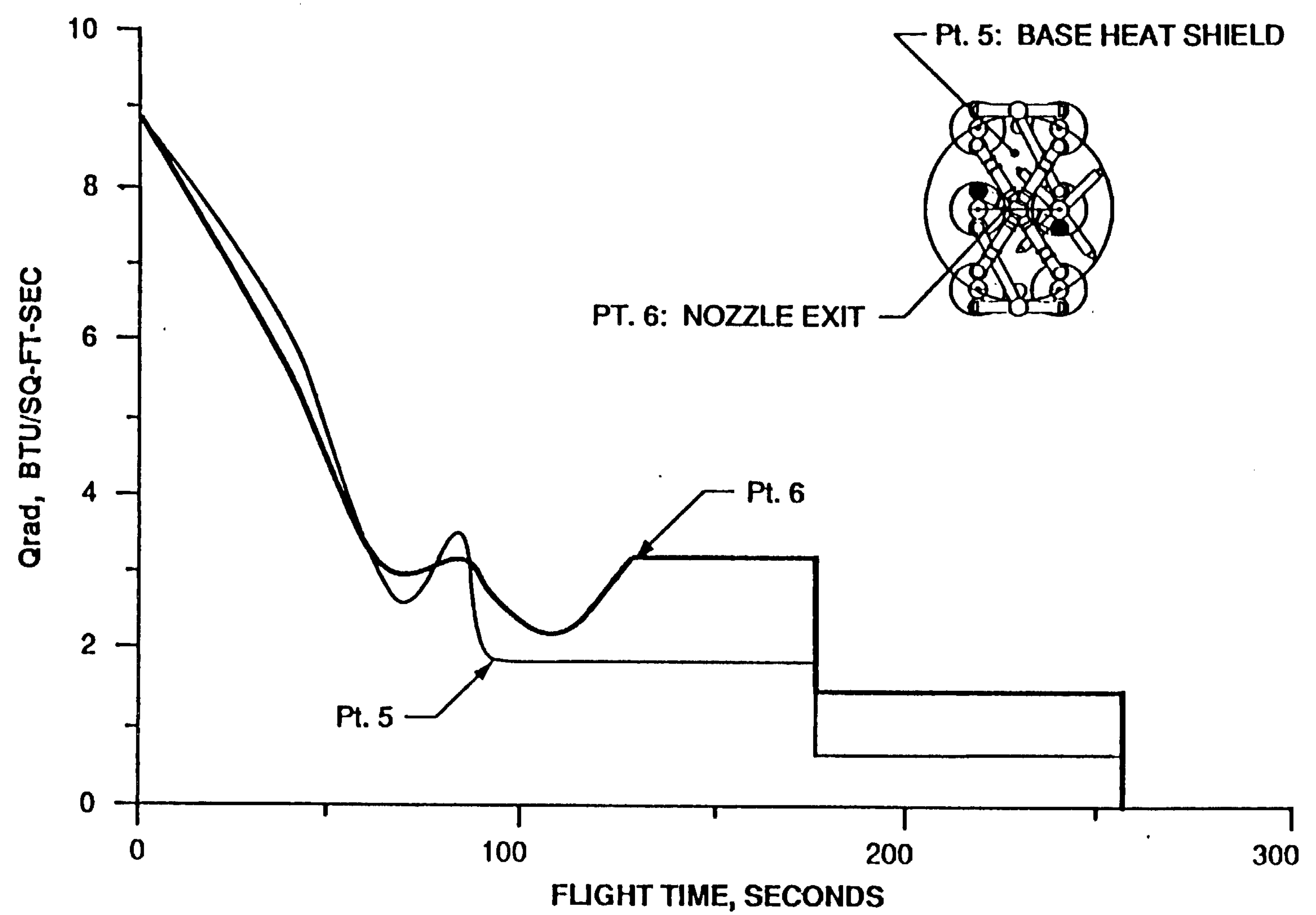
NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

1.5 STAGE REFERENCE



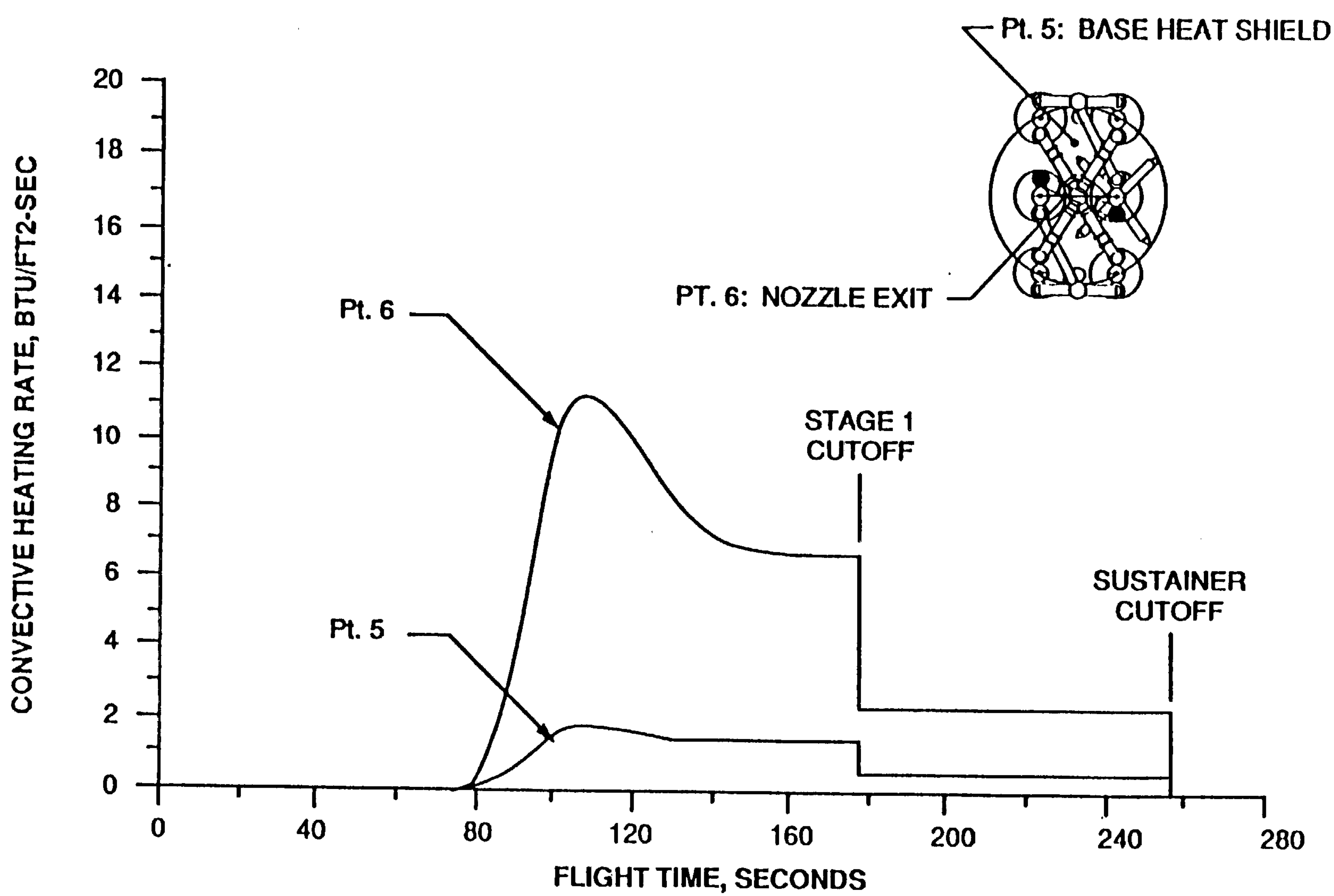
NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

1.5 STAGE SIX-PACK



NLS - AERO/THERMODYNAMIC PANEL - VIFM-2
TASK 3-FM-006 ASCENT PLUME INDUCED ENVIRONMENTS

1.5 STAGE SIX-PACK



CONCLUSIONS

- ***CONCLUSIONS***
  - HLLV more severe than 1.5 Stage because
    - ASRB radiation
    - ASRB recirculation around STME shroud and over core base region
  - 1.5 Stage reference has higher heating to interior STME nozzle than 1.5 Stage 6–pack
  - Plume Induced Flow Separation (PIFS) region may extend forward to mid booster for HLLV
- ***ISSUES FOR FOCUSED ANALYSIS DURING FALL 1991***
  - ASRB radiation to STME nozzle interior
  - H_2 film cooling and by-pass injectants on STME plume afterburning
  - Gimbaling of HLLV ASRB toward STME
  - Jettisoning of booster module engines while sustainer engines firing
  - PIFS as ignition source for H_2 vent gases

REMTECH

NLS

BASE HEATING/BASE BURNING
STME TURBINE EXHAUST DISPOSAL REVIEW

NOVEMBER 4, 1991

PREPARED BY:
ROBERT L. BENDER
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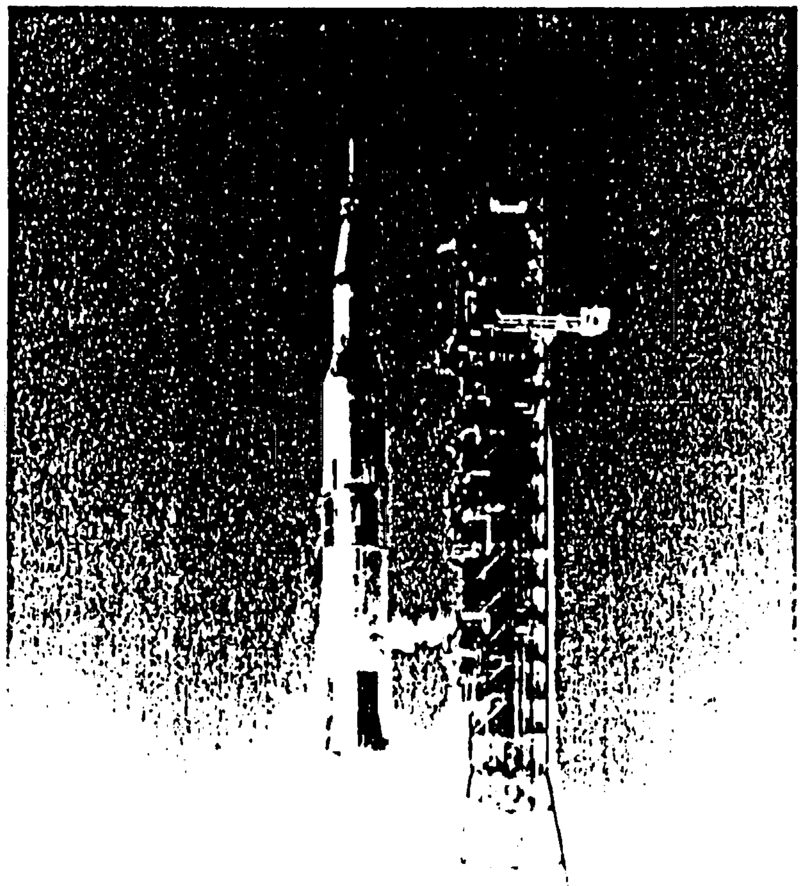
OUTLINE

BACKGROUND

REMTECH

TYPICAL LAUNCH VEHICLE PLUMES AND BASE FLOWFIELDS

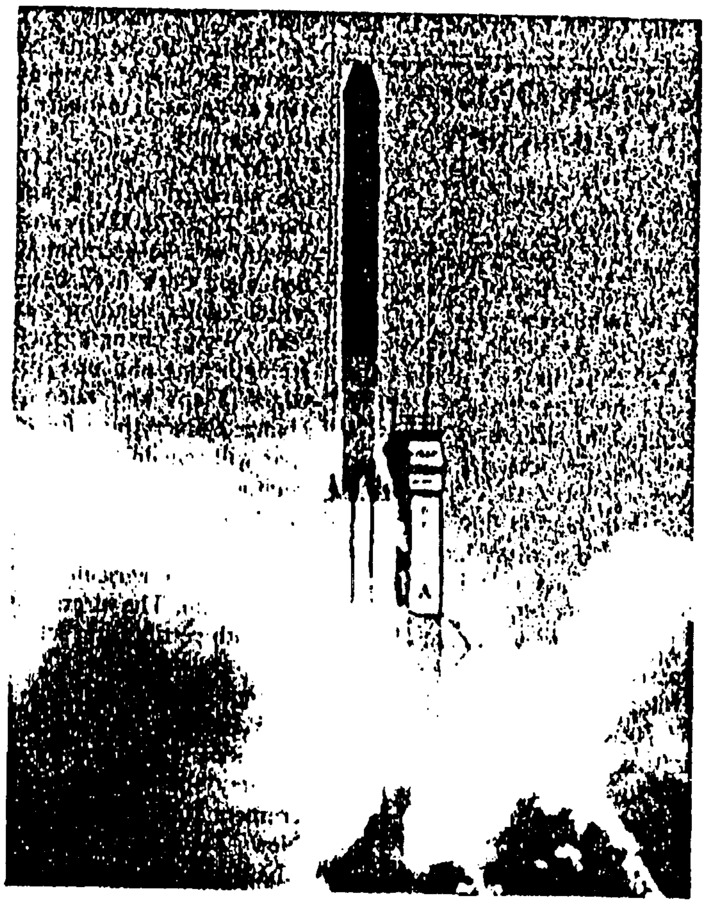


Saturn V

Atlas

BACKGROUND

TYPICAL LAUNCH VEHICLE PLUMES AND BASE FLOWFIELDS



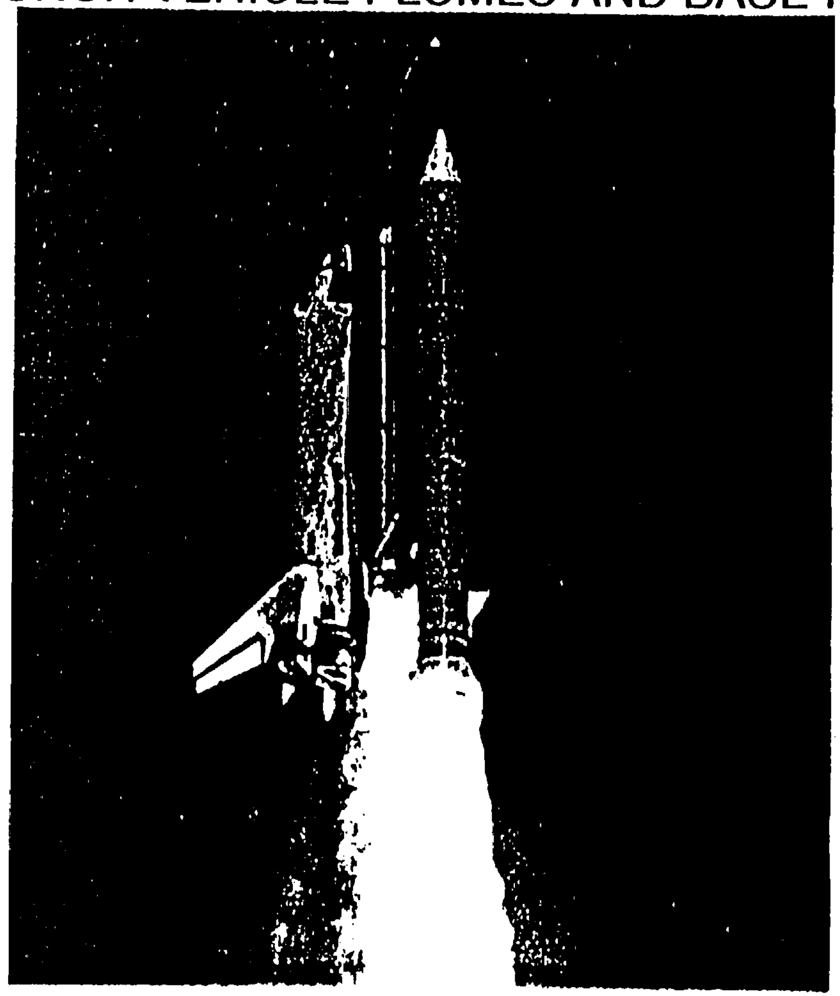
Delta

Titan

BACKGROUND

REMTECH

TYPICAL LAUNCH VEHICLE PLUMES AND BASE FLOWFIELDS



NSTS Space Shuttle

BASE HEATING ENVIRONMENT COMPONENTS

REMTECH

The base heating environment is composed of a convective heating component and radiation component. Convection occurs as the base region gases flow over the base structure. Radiation to the base may be the combined radiation from several sources including: the core of the downstream plumes, the plume mixing boundaries, plume interaction regions, local hot gases in the base, localized burning in the base, or, occasionally , from other hot structures in the base. Most analysts are concerned with main plume radiation and convective heating from reversed gases.

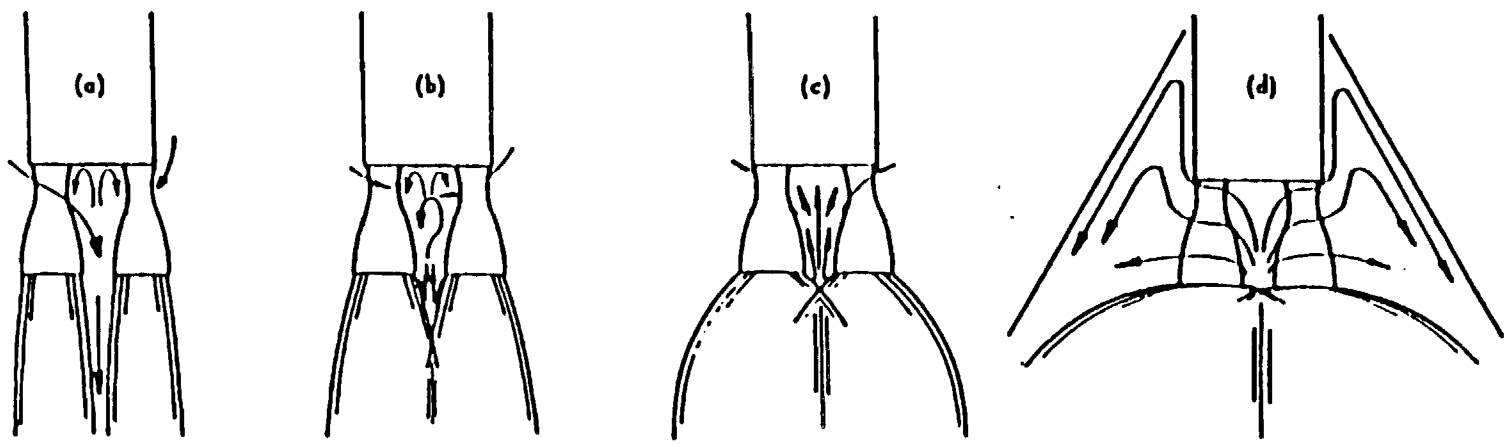
RADIATION SOURCES

- LOW ALTITUDE (< 70 kft)
  - * Plume Core (Mach Disk)
  - * Afterburning
  - * Baseburning (Turbine Exhaust)
- HIGH ALTITUDE (> 70 kft)
  - * Plume Core (Near Field)
  - * Plume Interaction Zones
  - * Base Recirculation
- SRM SHUTDOWN SPIKE

CONVECTION SOURCES

- COOLING FROM AMBIENT AIR
- HEATING FROM RECIRCULATED PLUME GASES
  - * PLUME-PLUME INTERACTIONS
  - * PLUME-FREESTREAM INTERACTIONS
- BASE BURNING FROM RECIRCULATED TURBINE EXHAUST

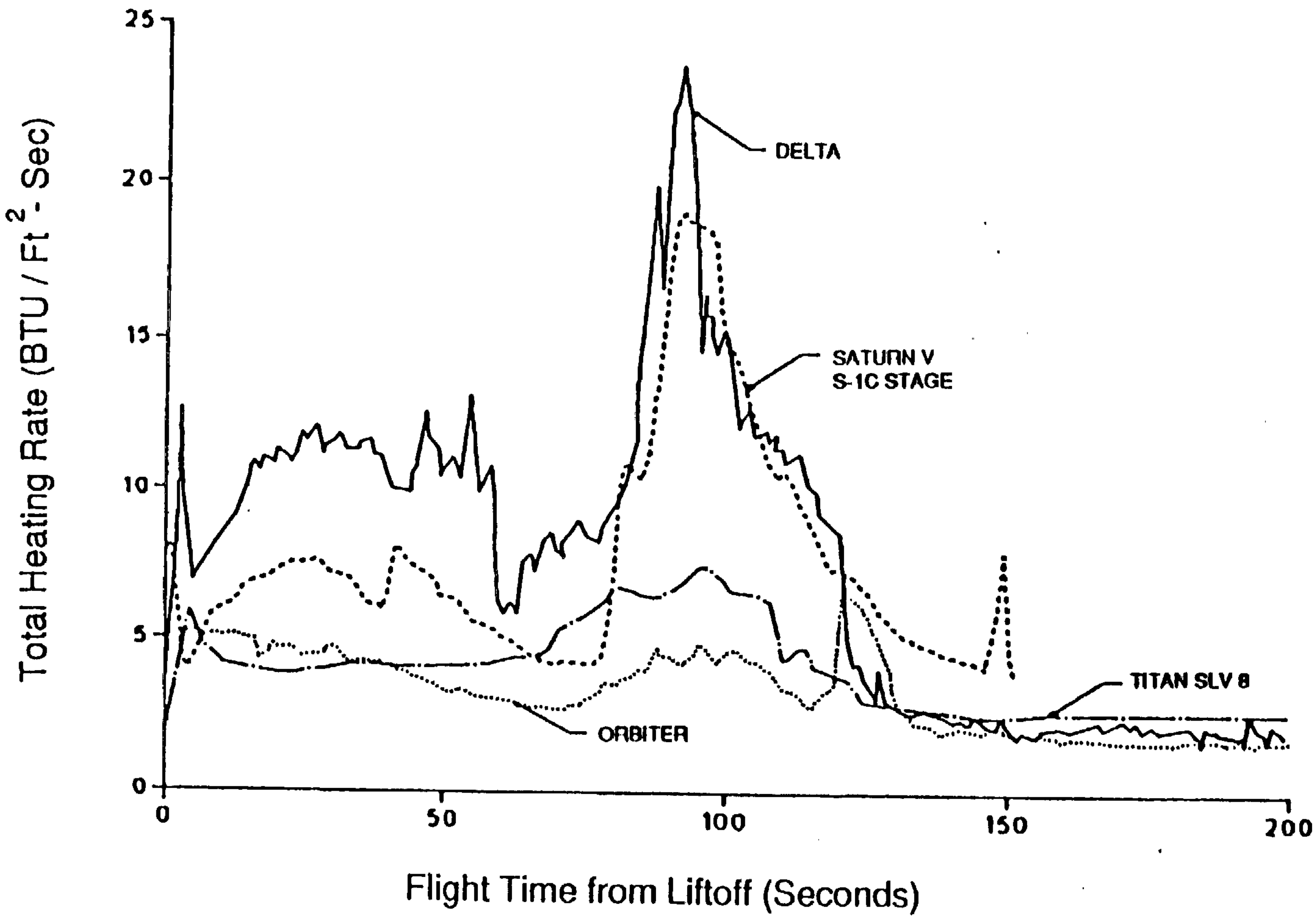
MULTINOZZLE ROCKET BASE FLOW PATTERNS



(a)	(b)	(c)	(d)
Low Altitude, $P_j \approx P_\infty$.	Medium Altitude, $P_j > P_\infty$.	High Altitude, $P_j >> P_\infty$.	High Altitude, $P_j >> P_\infty$.
No Jet Interference.	Some Jet Interference.	Severe Jet Interference.	Severe Jet Interference.
	Minor Jet Recirculation.	Extensive Jet Recirculation.	Plume-Induced Flow Separation.

BACKGROUND

FLIGHT BASE HEATING RATES FROM U.S. LAUNCH VEHICLES
TYPICAL BASE HEAT SHIELD DATA

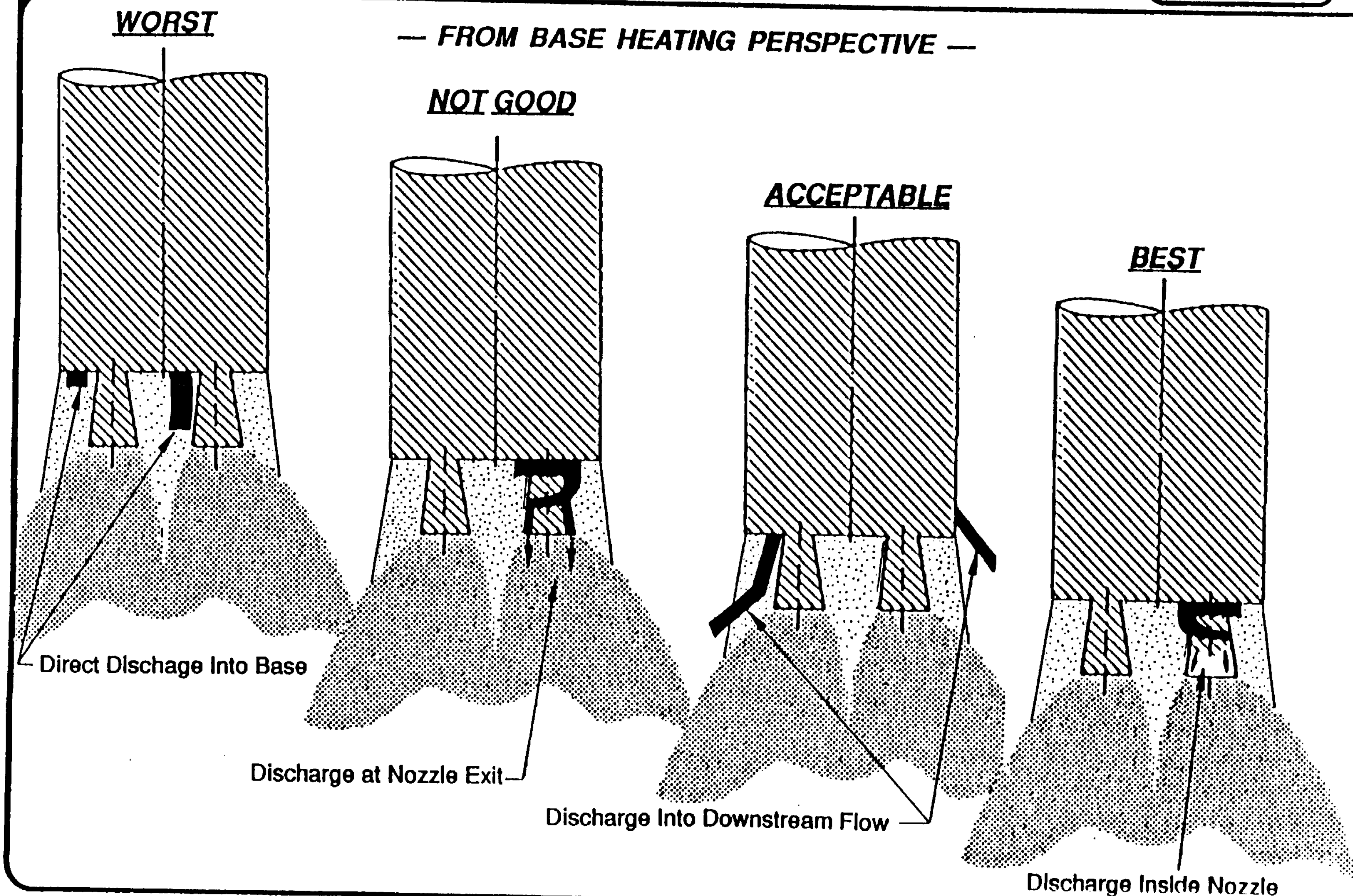


HOW DOES TURBINE EXHAUST DISPOSAL AFFECT BASE HEATING?

- If turbine exhaust dumped outboard or downstream
  - Combustible gases will burn in downstream plume and are not entrained in local recirculation pattern.
  - Amount of combustible exhaust product in engine nozzle boundary layer is small — so base region convection due to recirculated gases is determined by nozzle boundary layer gas temperature.
  - Afterburning in near plume and resultant change in plume radiation is minimized.
- If turbine exhaust dumped directly in base, engine nozzle, or nozzle exit plane.
  - Local combustion of turbine exhaust gases will occur in base region when oxidizer is present and base pressure is sufficient — referred to as base burning.
  - Base burning increases base gas temperature, alters base flow patterns, and may dramatically increase base region convection and local gas radiation.
  - Nozzle injection and subsequent afterburning changes plume radiation characteristics, often increasing downstream plume radiation.

PREFERRED TURBINE EXHAUST DISPOSAL SCHEMES

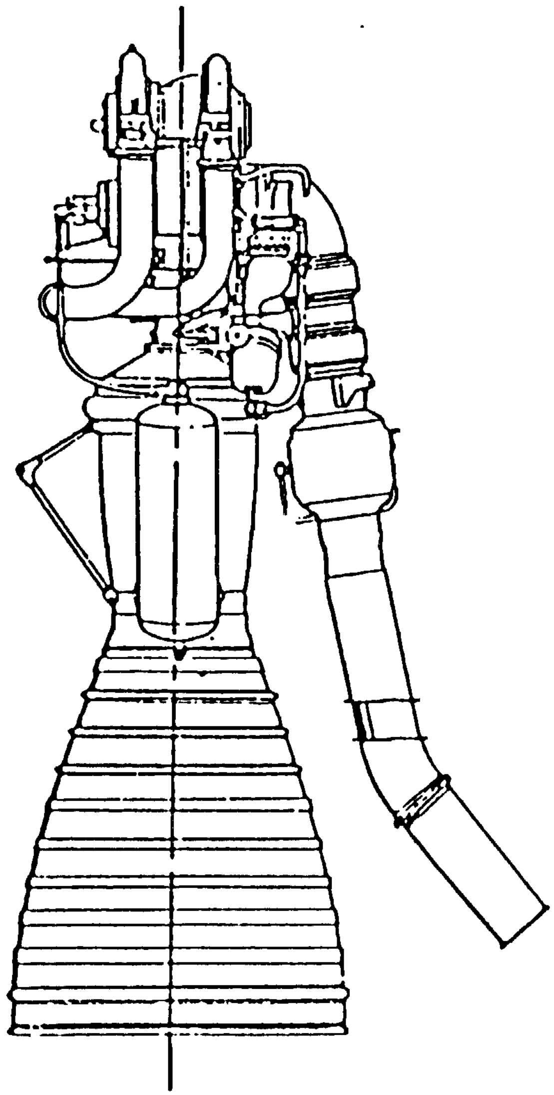
— FROM BASE HEATING PERSPECTIVE —



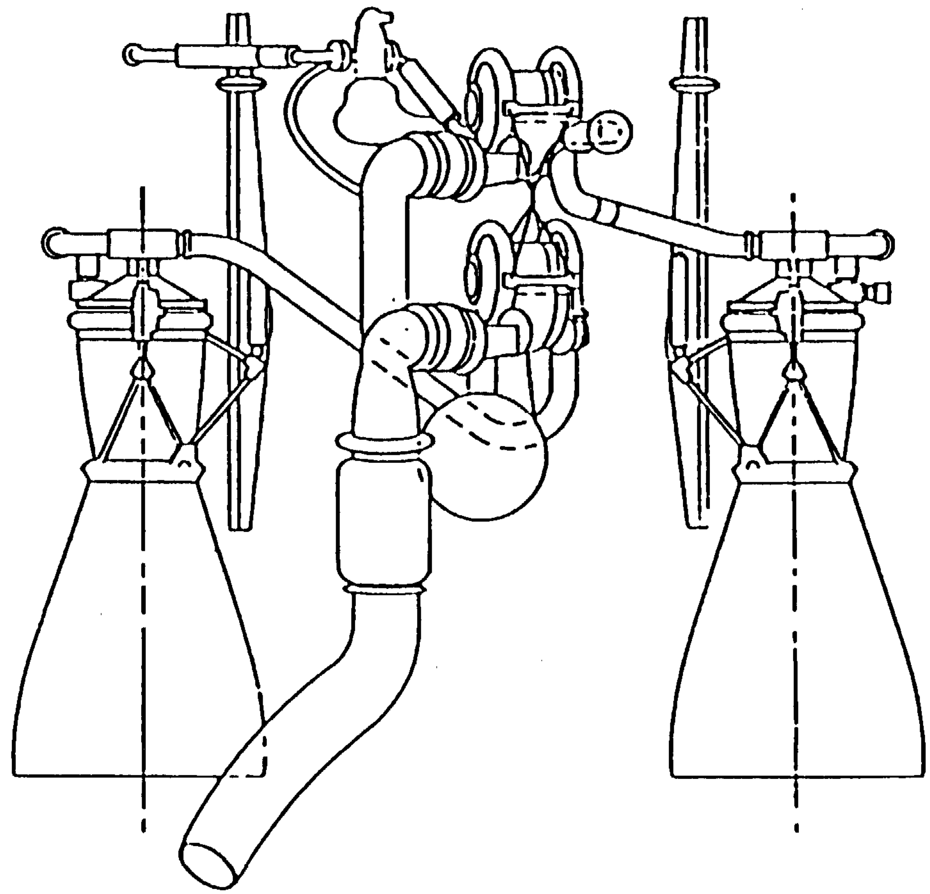
PAST EXPERIENCE WITH TURBINE EXHAUST DISPOSAL
--- LARGE U.S. LAUNCH VEHICLES ---

VEHICLE	T.E. DISPOSAL SCHEME	EXPERIENCE/LESSON LEARNED
JUPITER -1A	• Duct Along Nozzle to Exit Plane • Change to Outboard Duct	• 1st Flight Failed Due to Base Heating • No failure
ATLAS	• Duct into Base - By Center Engine • Change to Outboard Duct	• 1st 2 Flights Failed Due to Base Heating • No Failure
DELTA	• Duct through Heat Shield	• High local heating on heat shield while SRM's attached
TITAN II	• Two ducts exiting slightly aft of boattail base. • Strong air scooping eliminates base burning.	• Heating not severe • No failure due to T.E. burning
TITAN III (Core)	• Core engine ignited at $H \geq 100$ kft; above altitude of serious burning.	• No trouble
SATURN I	• Inbd engine ducted to fin outbd of base • Outbd engine into nozzle through exhausterator.	• High heating early in flight • No failure due to T.E. burning
SATURN IB	• Inbd engine ducted through 4 crescent opening in flame shield • Exhausterator on outbd engine	• T.E. exhaust did not burn; cooled flame shield • No failure
SATURN V	S-IC Stage — F-1 Engine T.E. Dumped in Nozzle @ $A/A^*=10$	• No Failure Due to Base Heating • Unburned RP-1 Afterburning in Plume @ Low Altitude, Burned in Base @ High Altitude
NSTS SPACE SHUTTLE	• No T.E. Disposal on SSME • SRB T.E. Dumped Outboard	• No Failure Due to Base Heating • Predictable Environments

TURBINE EXHAUST DISPOSAL THROUGH EXTERNAL DUCT

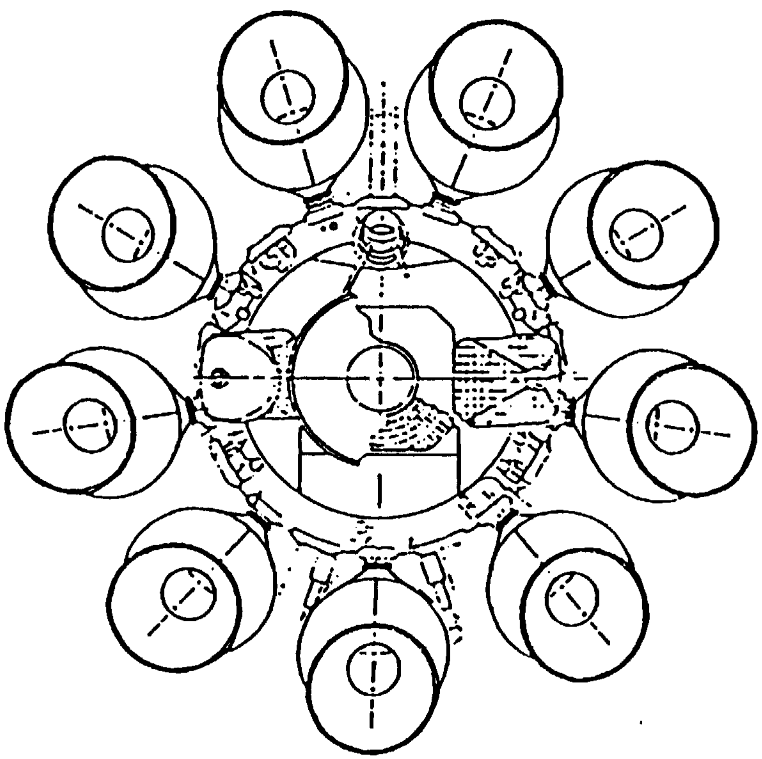


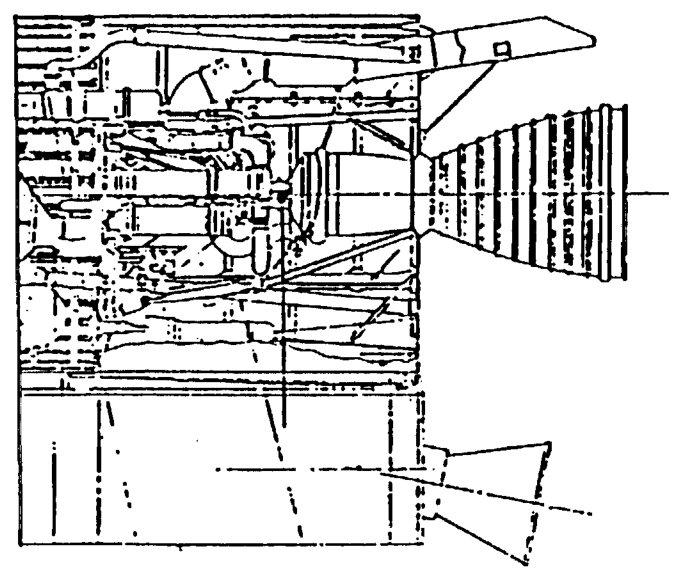
EARLY ATLAS MA-3 BOOSTER



LATER ATLAS MA-5 BOOSTER

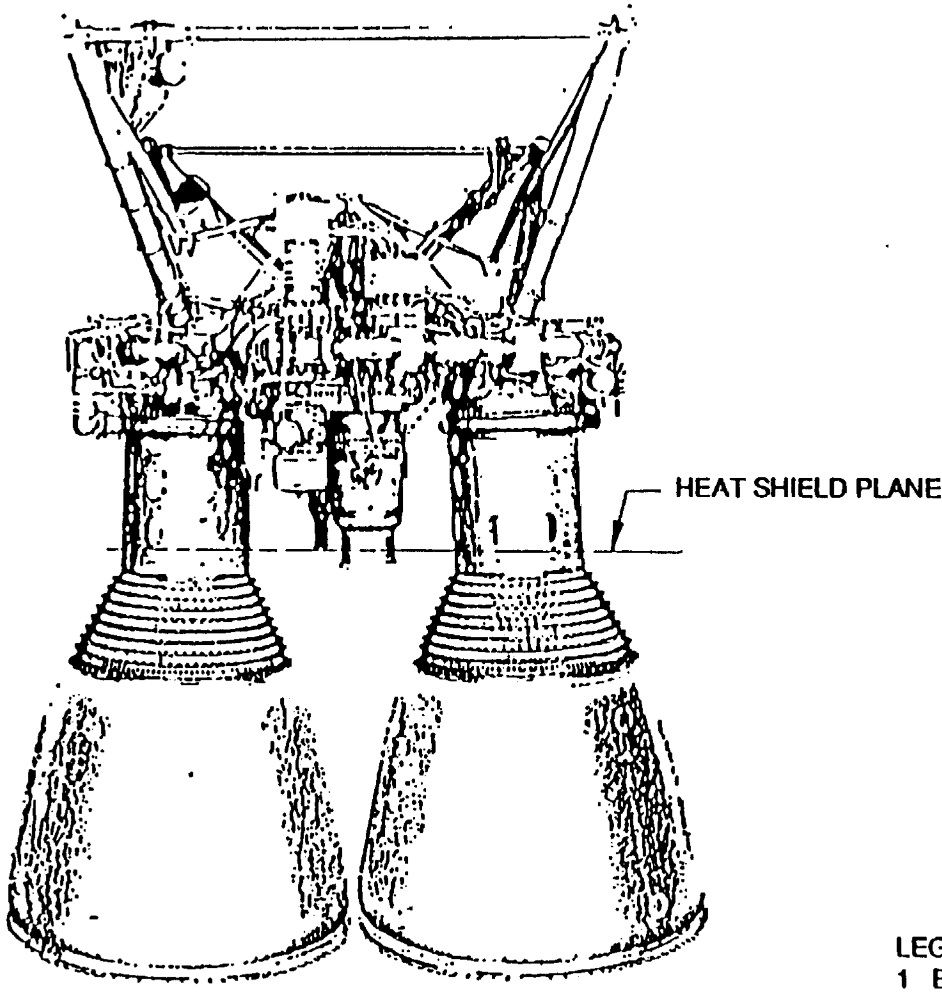
TURBINE EXHAUST DISPOSAL THROUGH DUCT PENETRATING HEAT SHIELD





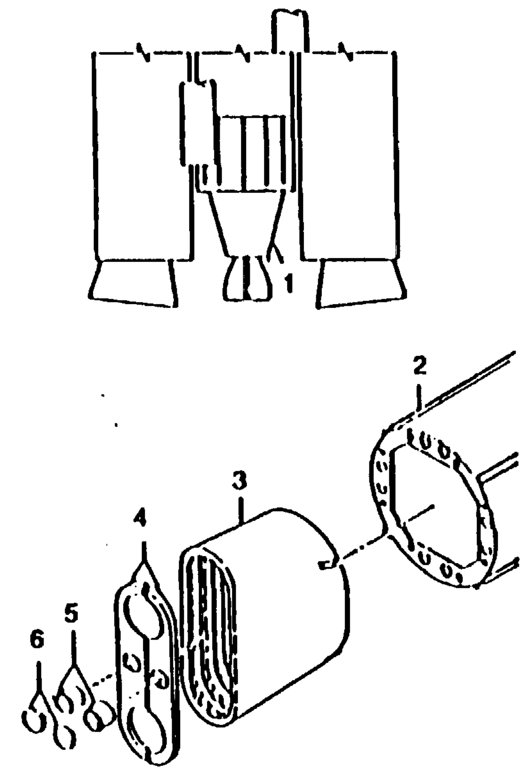
DELTA VEHICLE WITH RS-27 MAIN ENGINE

TURBINE EXHAUST DISPOSAL THROUGH BASE HEAT SHIELD



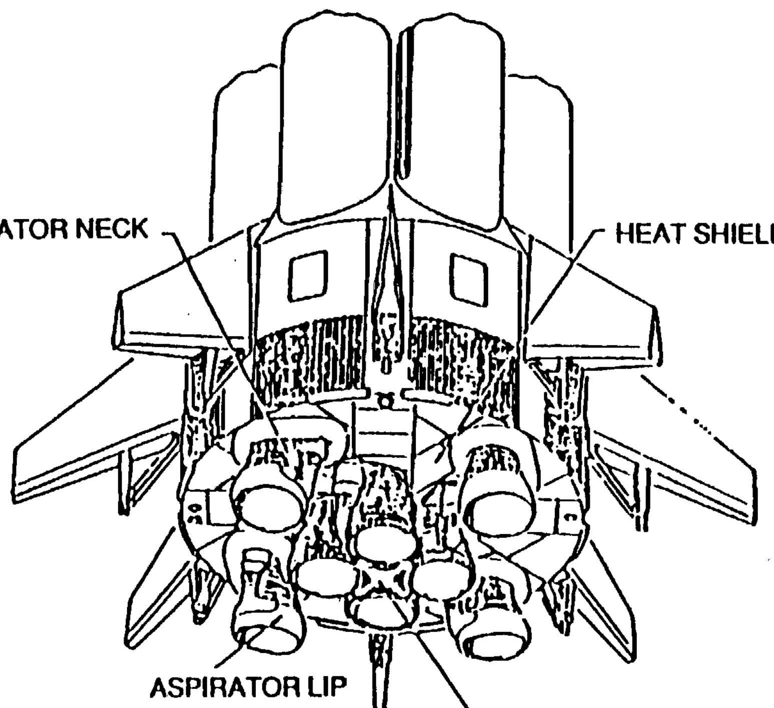
TITAN IV BOATTAIL HEAT SHIELD

- Engines ignited at altitude
- Closures over nozzles and T.E. ducts



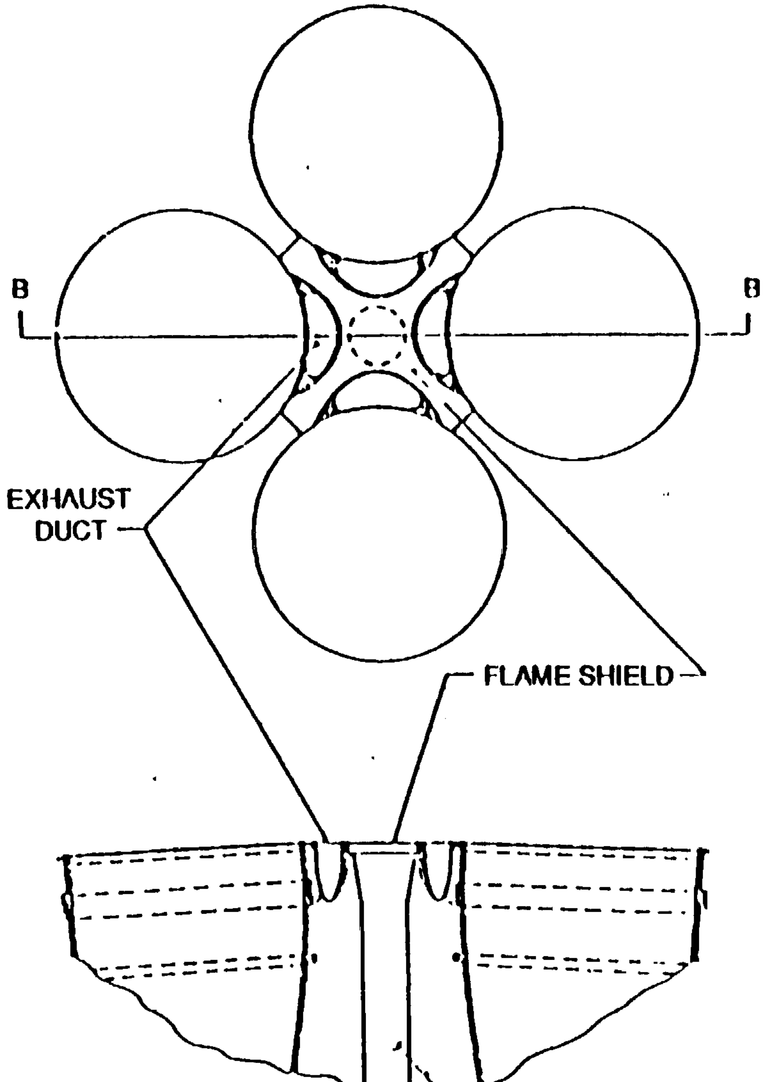
LEGEND:
1 BOATTAIL SHROUD
2 STAGE 1 AIRFRAME
3 HEAT SHIELD
4 HEAT SHIELD BULKHEAD
5 EXHAUST STACK AND START CARTRIDGE SHIELDS
6 EXHAUST STACK COVERS

TURBINE EXHAUST DISPOSAL THROUGH BASE HEAT SHIELD

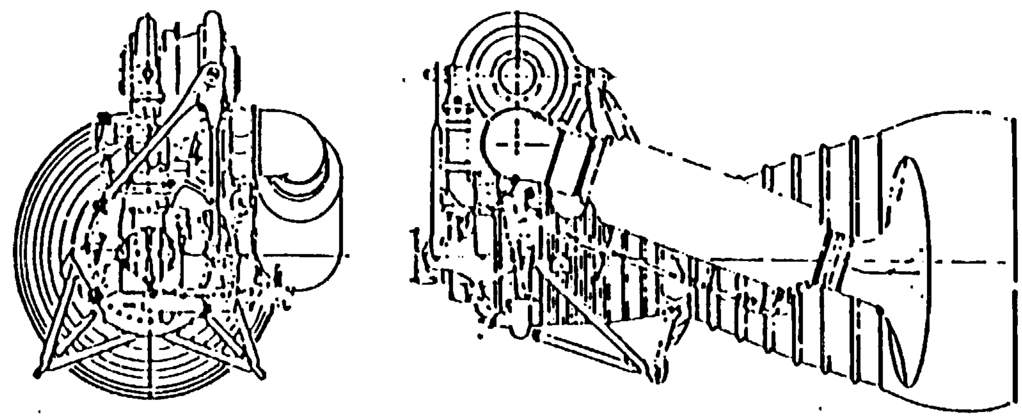


SATURN 1B BOOSTER

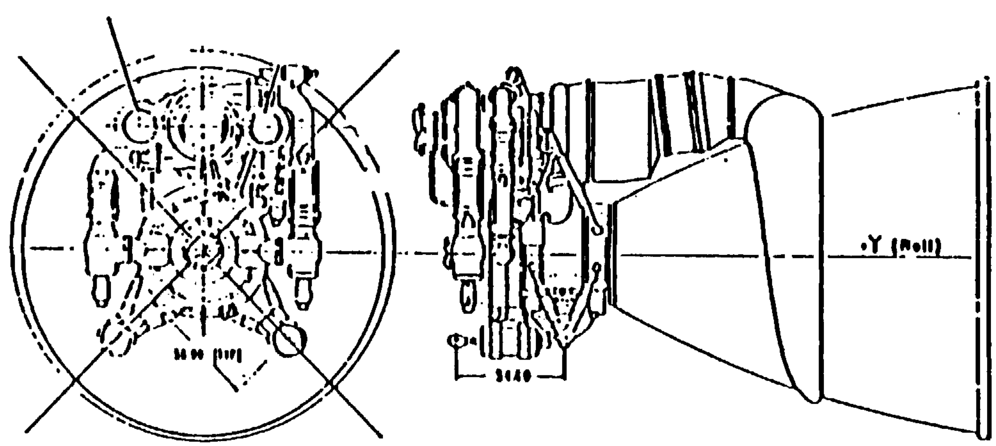
CENTER H-1 ENGINES - FLAME SHIELD



TURBINE EXHAUST DISPOSAL INSIDE NOZZLE



SATURN I AND IB BOOSTERS - OUTBOARD H-1 ENGINE



SATURN V/S-1C STAGE - F-1 ENGINE

SUMMARY OF TURBINE EXHAUST DISPOSAL FLIGHT EXPERIENCE

- Flight vehicles with turbine exhaust disposal into base, engine nozzle, or external flow.
  - ATLAS
  - SATURN 1 & 1B, 1st Stage
  - SATURN V, 1st Stage
  - DELTA

  } LO_2/RP-1 Propellants

  - TITAN — Aerozine 50/UDMH Propellants (Storable)
- Flight vehicles which utilized LO_2/LH_2 propellants.
  - S–IV Stage, SATURN 1
  - S–II Stage, SATURN V
  - S-IV B Stage, SATURN V

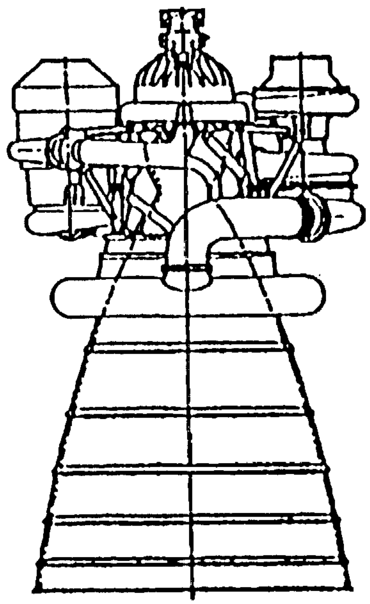
  } T.E. Dumped inside nozzle-high altitude.

  - Shuttle Orbiter — Regeneratively cooled nozzle — no T.E. Discharge

THE NLS - STME TURBINE EXHAUST DILEMMA

- The STME with film/convective dump cooled nozzle:
  - is a new concept, outside experience range
  - creates potential for large mass flow of low energy, unburned H_2 at nozzle exit lip
  - H_2 will burn over wide range of mixture ratios (and pressures) with oxygen (air) present in base.
- Both NLS configurations have complex base flowfields and potential for low altitude recirculation
  - HLLV — close proximity of ASRB (with skirt) and STME (with shroud).
  - 1.5 Stage — close proximity of sustainer engines and sustainer/booster engines

STME FILM/CONVECTIVE DUMP COOLED NOZZLE



Thrust, lbs	683,000
Chamber Pressure, psia	2250
Mixture Ratio	6.0
Min. Specific Impulse (vac) sec	430.5
Weight, lbs	8000
Area Ratio	45

MAIN CHAMBER

$P_o - 2250^\circ\ psia$

$T_o - 6708^\circ\ R$

$\dot{\omega} - 1292.7\ lbm/sec$

TURBINE EXHAUST DISCHARGE

•Primary Film Coolant

$P_o - 204\ psia$

$T_o - 1190^\circ\ R$

$\dot{\omega} - 24.4\ lbm/sec$

•Secondary Film Coolant

$P_o - 80.3\ psia$

$T_o - 1190\ R$

$\dot{\omega} - 4.26\ lbm/sec$

•Convective Coolant

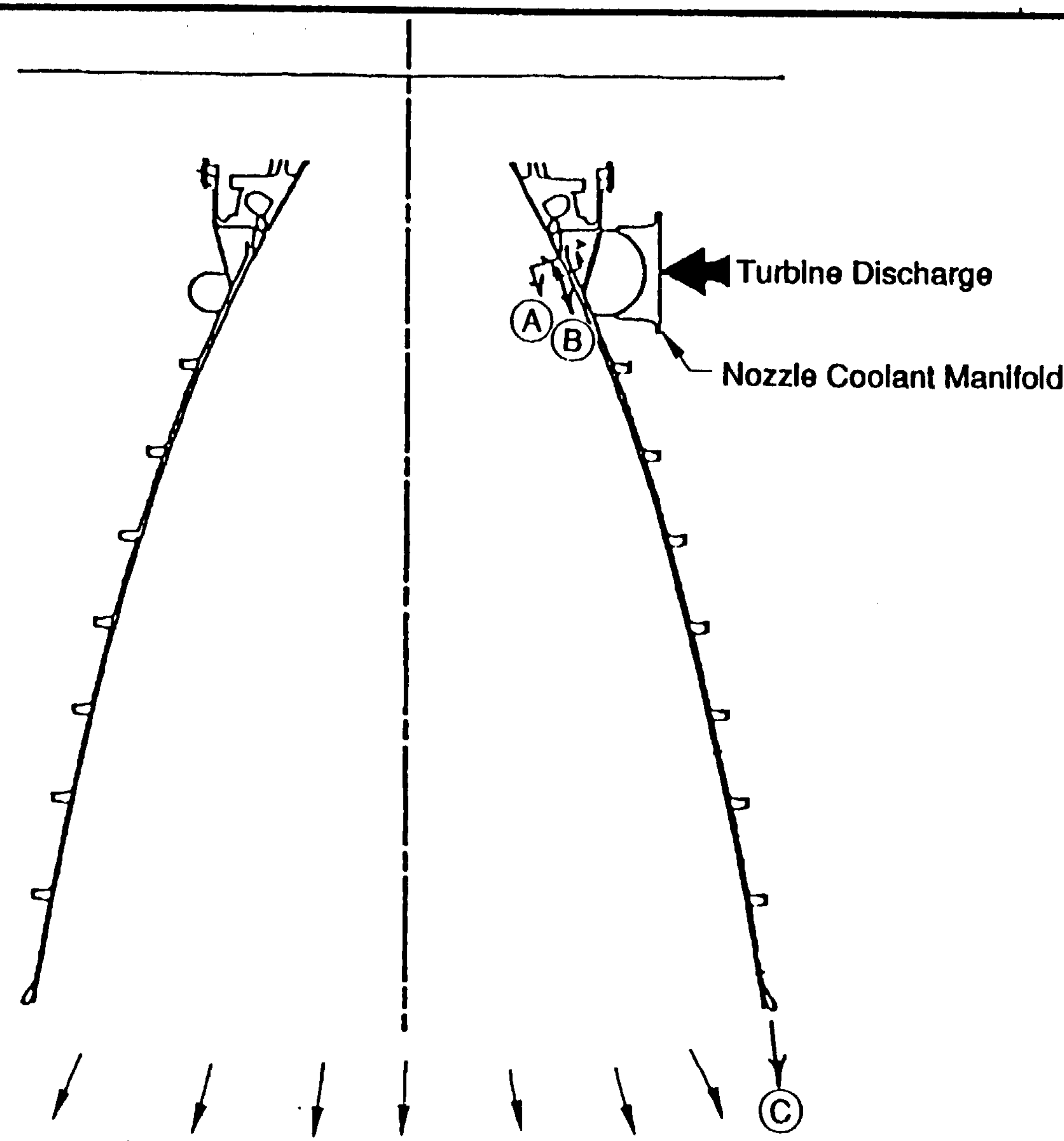
$P_o - 88.8\ psia$

$T_o - 1462.4^\circ\ R$

$\dot{\omega} - 35.4\ lbm/sec$

NOTE: Turbine exhaust is: 47% H_2
53% H_2O (Steam)

STME HYDROGEN FLOW RATES



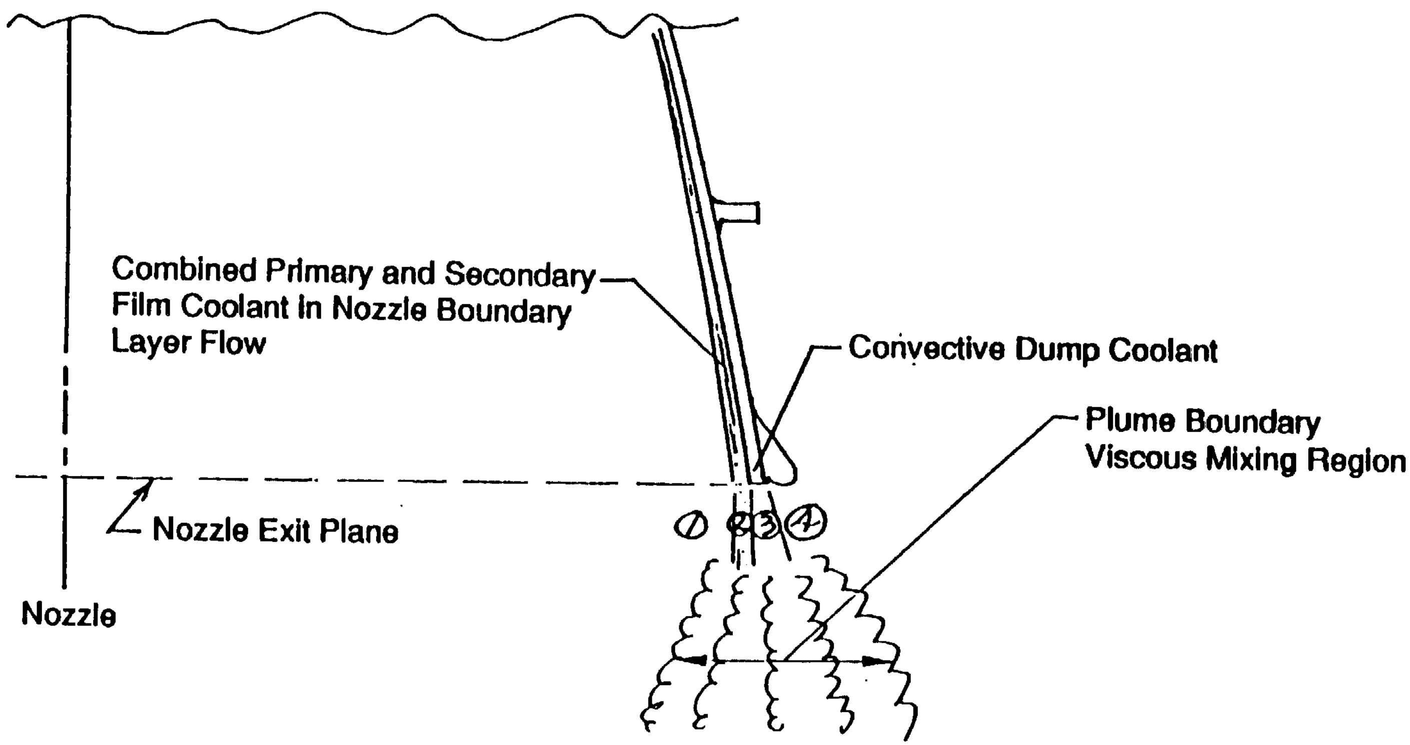
(A) ***Secondary Film Coolant***
2.0 lbm H_2/sec.

(B) ***Primary Film Coolant***
11.4 lbm H_2/sec.

(C) ***Convective Dump Coolant***
16.5 lbm H_2/sec.

MAIN ENGINE FLOW: $16.7\,lbm\,H_2/sec\,@\,O/F = 7.1$

STME PLUME EXPANSION/RECIRCULATION FLOWFIELD



<u>Four (4) Stream Mixing Problem</u>

1) Nozzle Inviscid Flow - $P_o \approx 2200$ psia
2) Film Coolant/Nozzle Boundary Layer Flow - $P_o \approx 200$ psia
3) Convective Dump Coolant Flow - $P_o \approx 90$ psia
4) Freestream or Base Region Flow - $P_o \approx 14.7$ or less psia

STME - H_2 RECIRCULATION POTENTIAL

- STME Without Film/Dump Cooled Nozzle
  - Based upon similar SSME analyses —
    - @ choking conditions, mass balance satisfied with 1% of boundary layer flow, which is 3 to 5% of total nozzle flow.
    - This translates to $\approx$ 0.25 lbm/sec of H_2 available to recirculate if STME nozzle is regen cooled.
- STME With Film/Dump Cooled Nozzle
  - Worse case assumes all low energy, low momentum film and dump coolant H_2 unable to penetrate high pressure shock regions of plume and recirculated to low pressure base region.
  - As much as 30 lbm/sec of H_2 available to burn in base region.

Note: This is $\approx$ 120 to 1 increase in H_2 available to burn!

BASE GAS TEMPERATURE WITH BURNING HYDROGEN

EXAMPLE COMPUTATION AT 20,000 FT. ALTITUDE

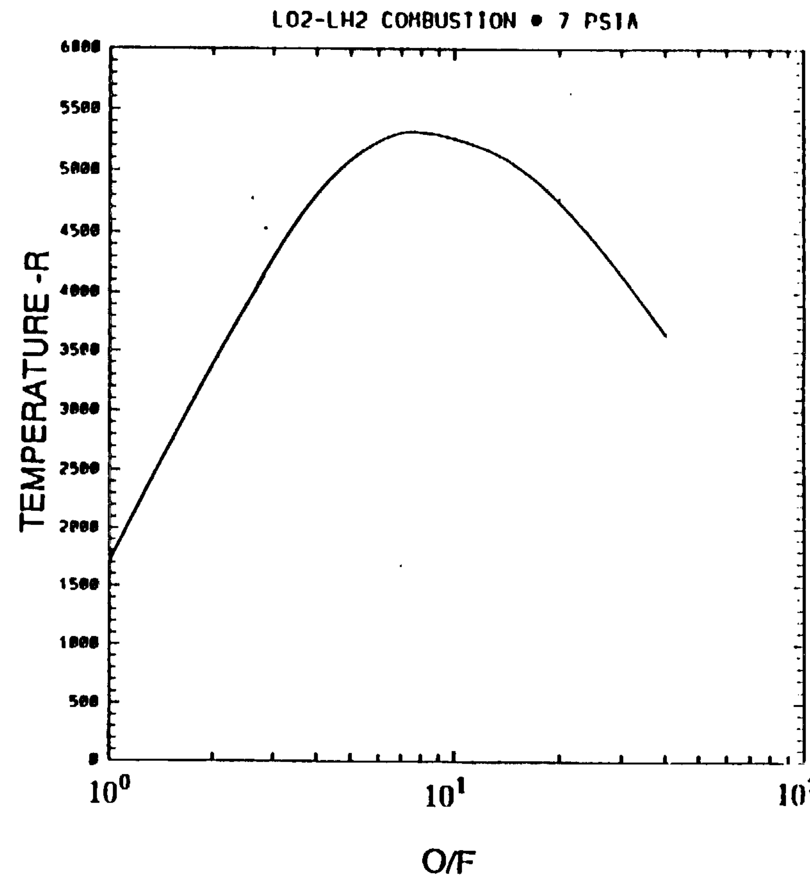
P_{BASE} = 7 psia

Available O_2 : 40 lbm (From air in total base volume)

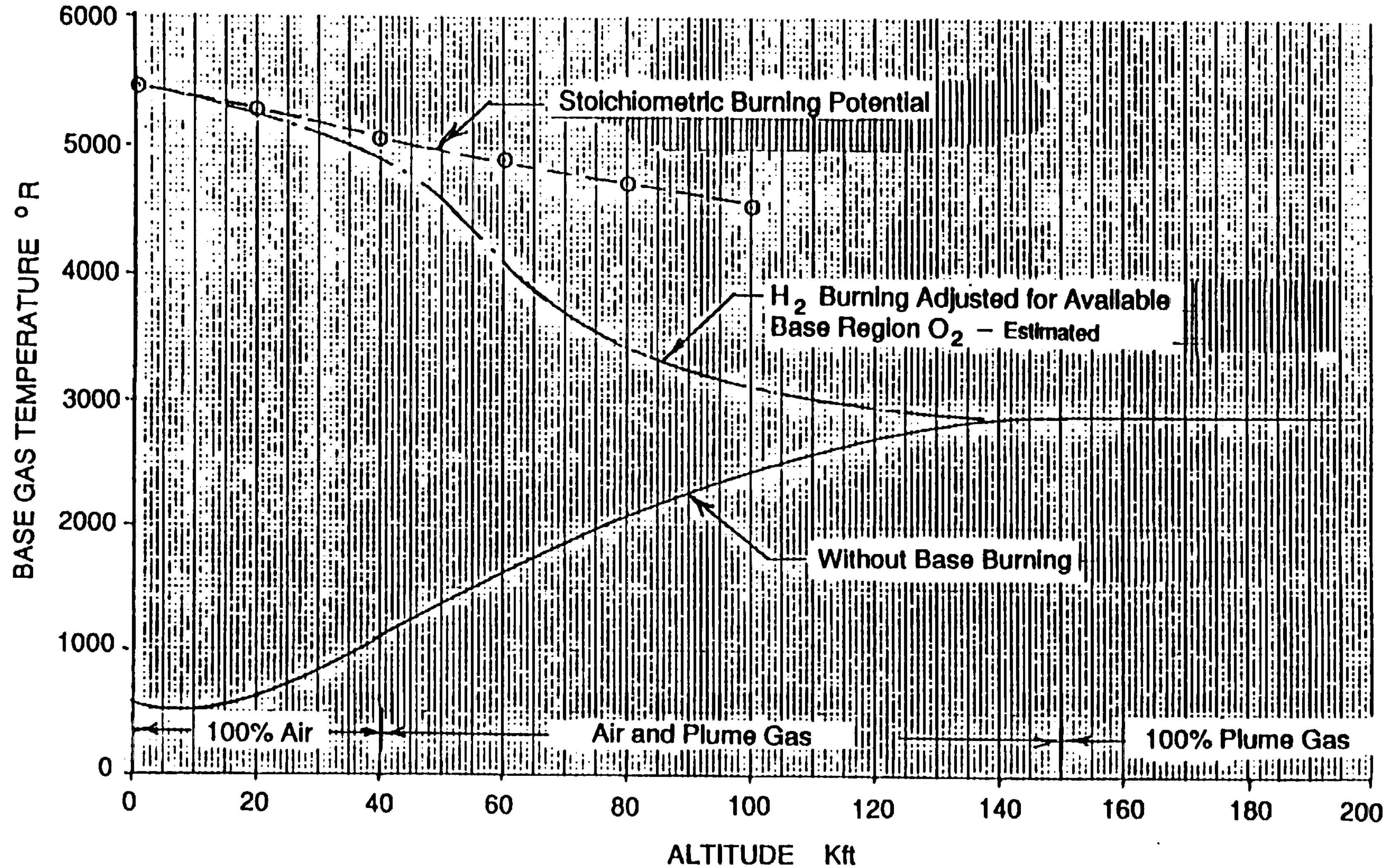
Available H_2 : 30 lbm (Per STME - all T.E.)
: 16.5 lbm (Per STME - By-pass only)

O/F	Assumption
8.0	Max Possible, depending on H_2 reversed and local flowfield
2.42	By-pass only - one engine fully mixed.
0.61	By-pass only - four engines fully mixed.
3.33	10% All T.E. reversed from four engines
22.2	1% all T.E. reversed from 1.5 stage six engines

Note: Assumes cold H_2



NLS ESTIMATED BASE GAS TEMPERATURE



NLS LOW ALTITUDE AIR-TURBINE EXHAUST COMBUSTION PRODUCTS - THERMODYNAMIC/TRANSPORT DATA

TYPICAL CEC OUTPUT

O/F = 15 P = 6.76 psia

	CHEMICAL FORMULA				WT FRACTION (SEE NOTE)	ENERGY CAL/MOL	STATE	TEMP DEG K	DENSITY G/CC
FUEL	H 2.00000				0.47000	0.000	G	298.15	0.0000
FUEL	H 2.00000	O 1.00000			0.53000	-57763.000	G	298.00	0.0000
OXIDANT	N 1.56170	O 0.41959	AR 0.00932	C 0.00030	1.00000	-28.200	G	298.15	0.0000

O/F= 15.0000 PERCENT FUEL= 6.2500 EQUIVALENCE RATIO= 1.0655 REACTANT DENSITY= 0.0000

THERMODYNAMIC PROPERTIES

P, ATM	0.4600
T, DEG K	2305
RHO, G/CC	5.7551-5
H, CAL/G	-107.1
S, CAL/(G)(K)	2.7627
M, MOL WT	23.666
(DLV/DLP)T	-1.00336
(DLV/DLT)P	1.0929
CP, CAL/(G)(K)	0.6441
GAMMA (S)	1.1797
SON VEL,M/SEC	977.5

MOLE FRACTIONS

AR	0.00714	CO	0.00007	CO2	0.00016	H	0.00250	H2	0.02949
H2O	0.35510	NO	0.00121	N2	0.59755	O	0.00029	OH	0.00504
O2	0.00145								

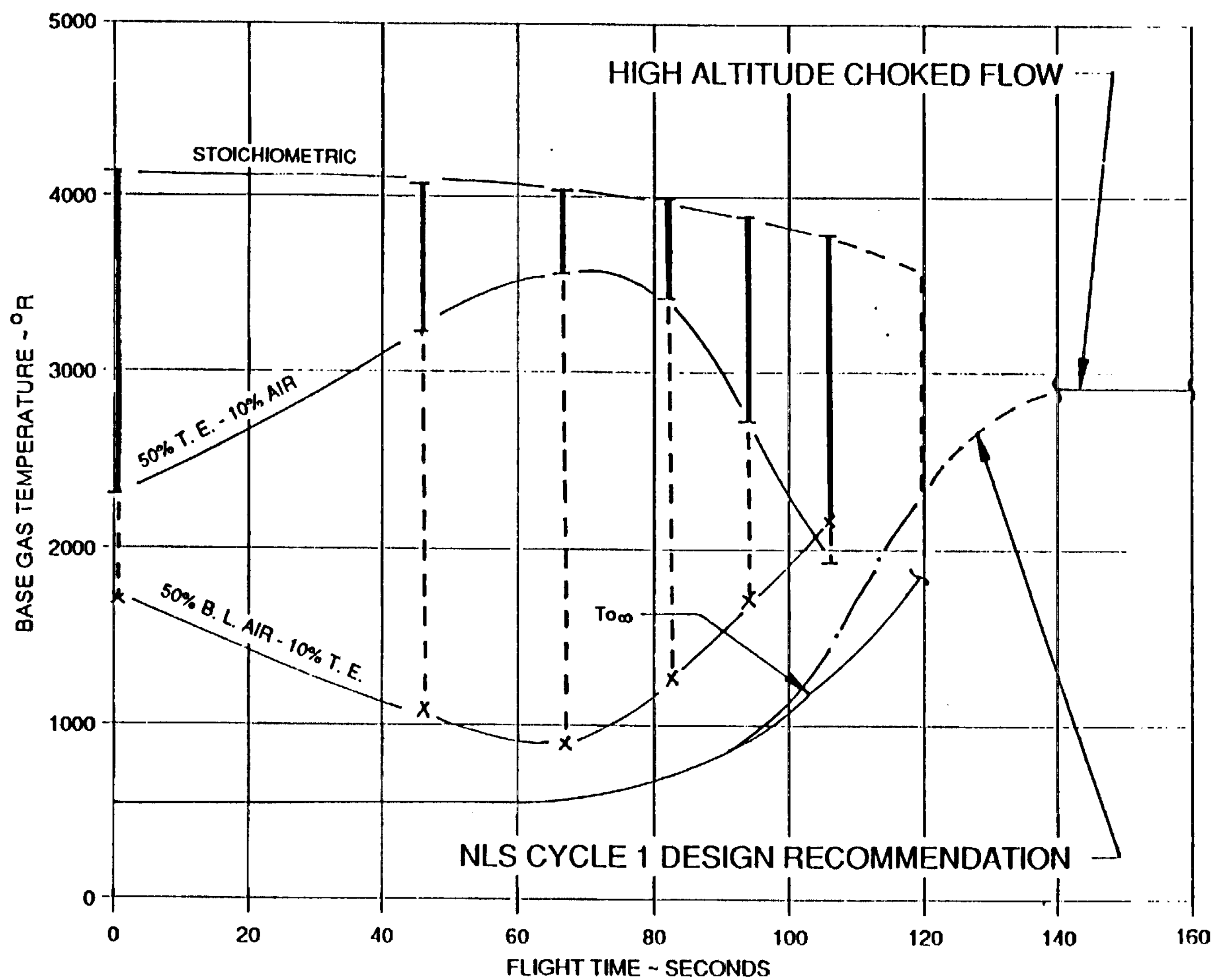
NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

TRANSPORT PROPERTIES AT ASSIGNED PRESSURES

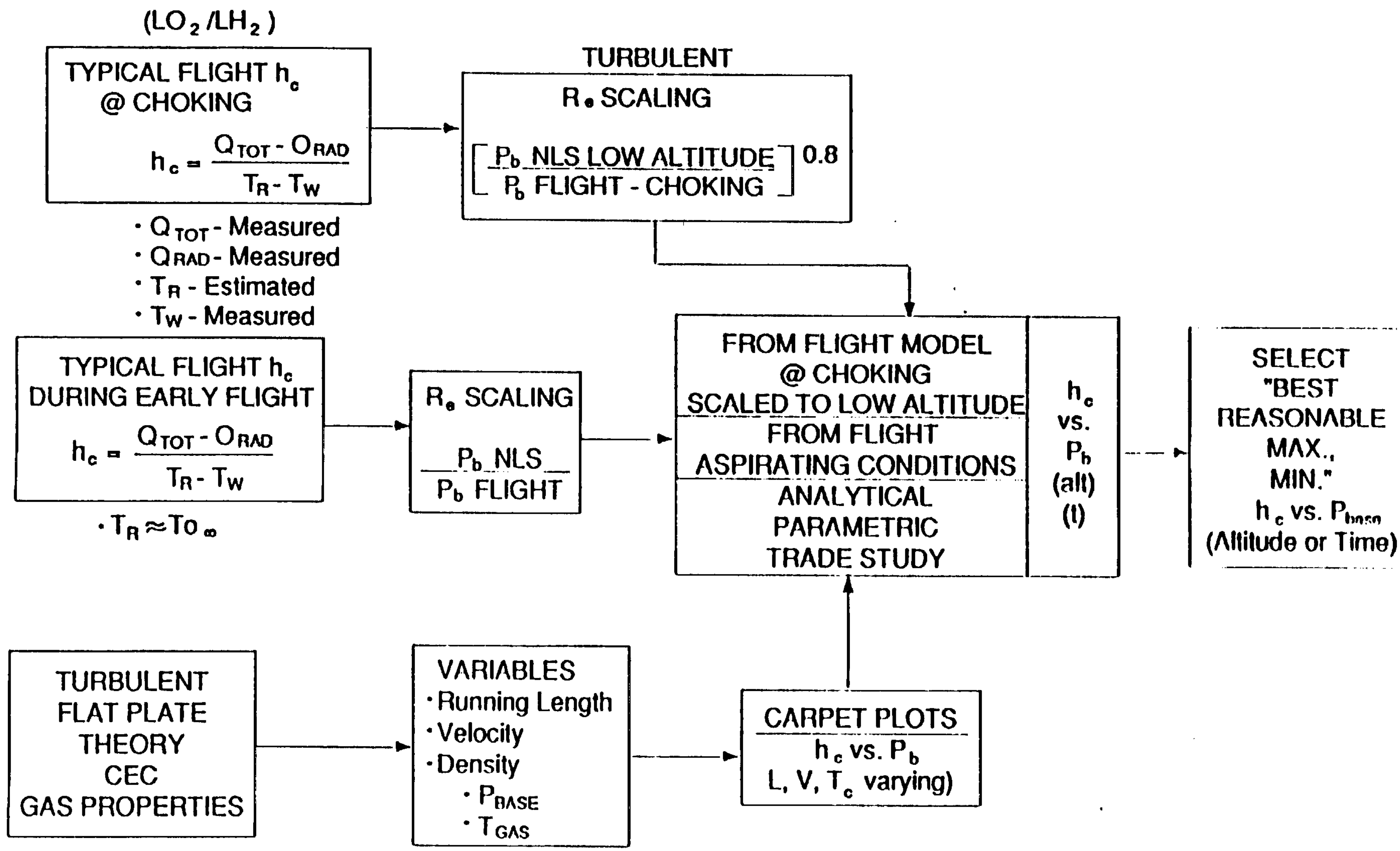
O/F= 15.0000 PERCENT FUEL= 6.2500 EQUIVALENCE RATIO= 1.0655 ENTHALPY= -107.1 CAL/G

TEMP	VISCOSITY	MONATOMIC COND	INTERNAL COND	FROZEN COND	REACTION COND	EQUILIBRIUM COND	CP FROZ	CP EQ	PRANDTL FROZ	PRANDTL EQ	LEWIS NUMBER
DEG K	POISE	CAL/(CM)(SEC)(K)					CAL/(G)(K)		DIMENSIONLESS		
2305	745.X10-6	246.X10-6	232.X10-6	477.X10-6	379.X10-6	857.X10-6	0.4250	0.6441	0.6640	0.5605	1.5433

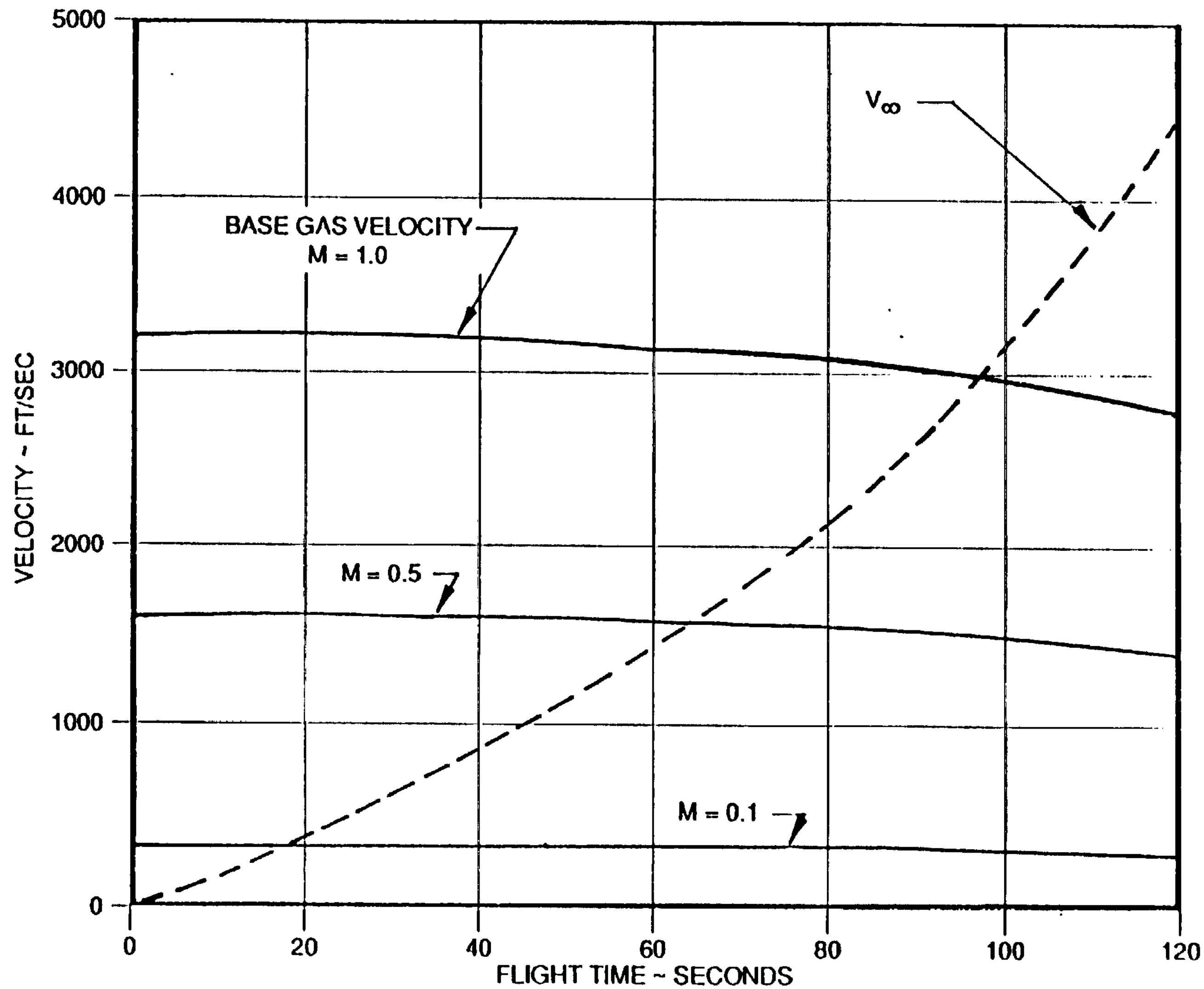
NLS - ESTIMATED CONVECTIVE BASE HEATING WITH TURBINE EXHAUST BASE BURNING



DETERMINATION OF CONVECTIVE HEAT TRANSFER COEFFICIENT

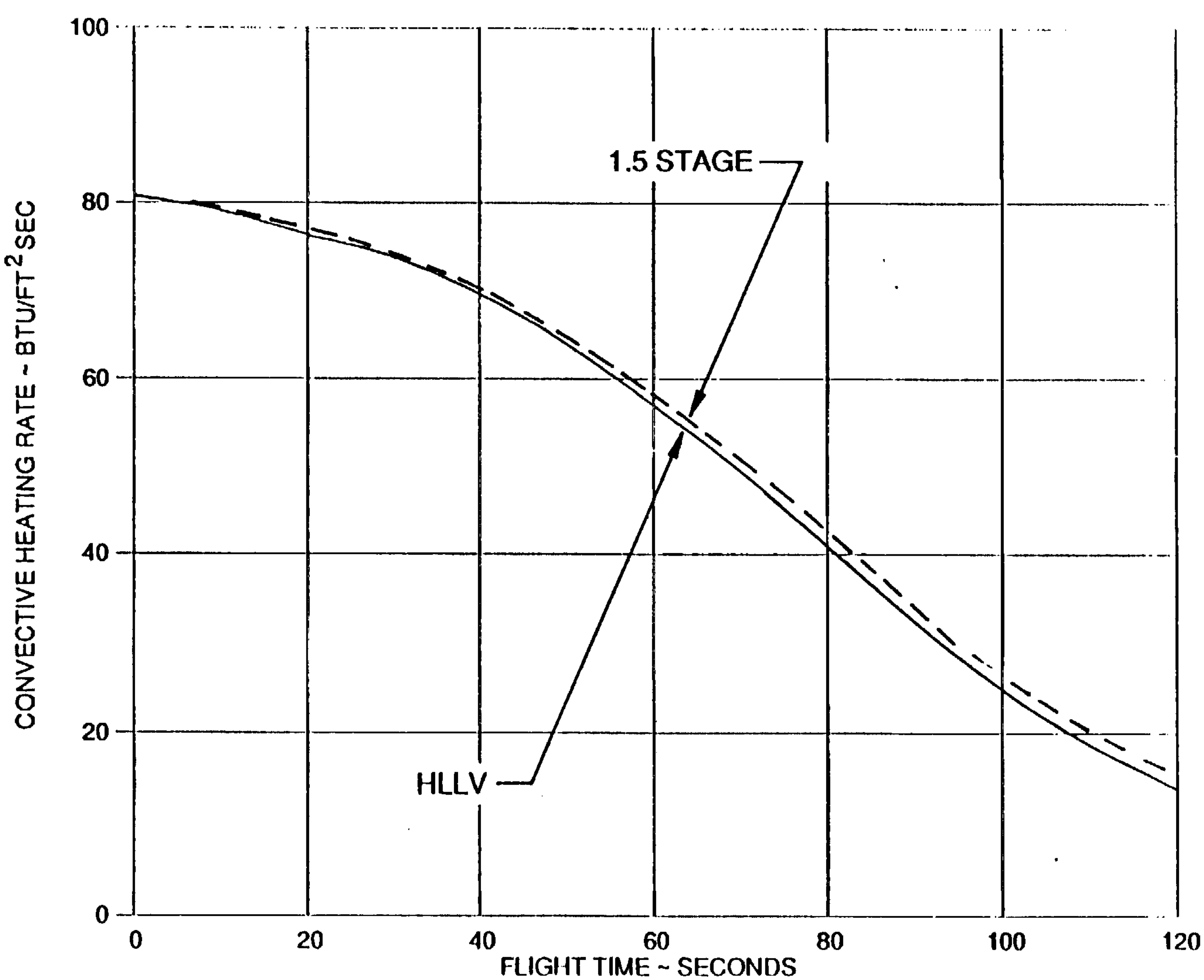


NLS BASE REGION VELOCITY ESTIMATES

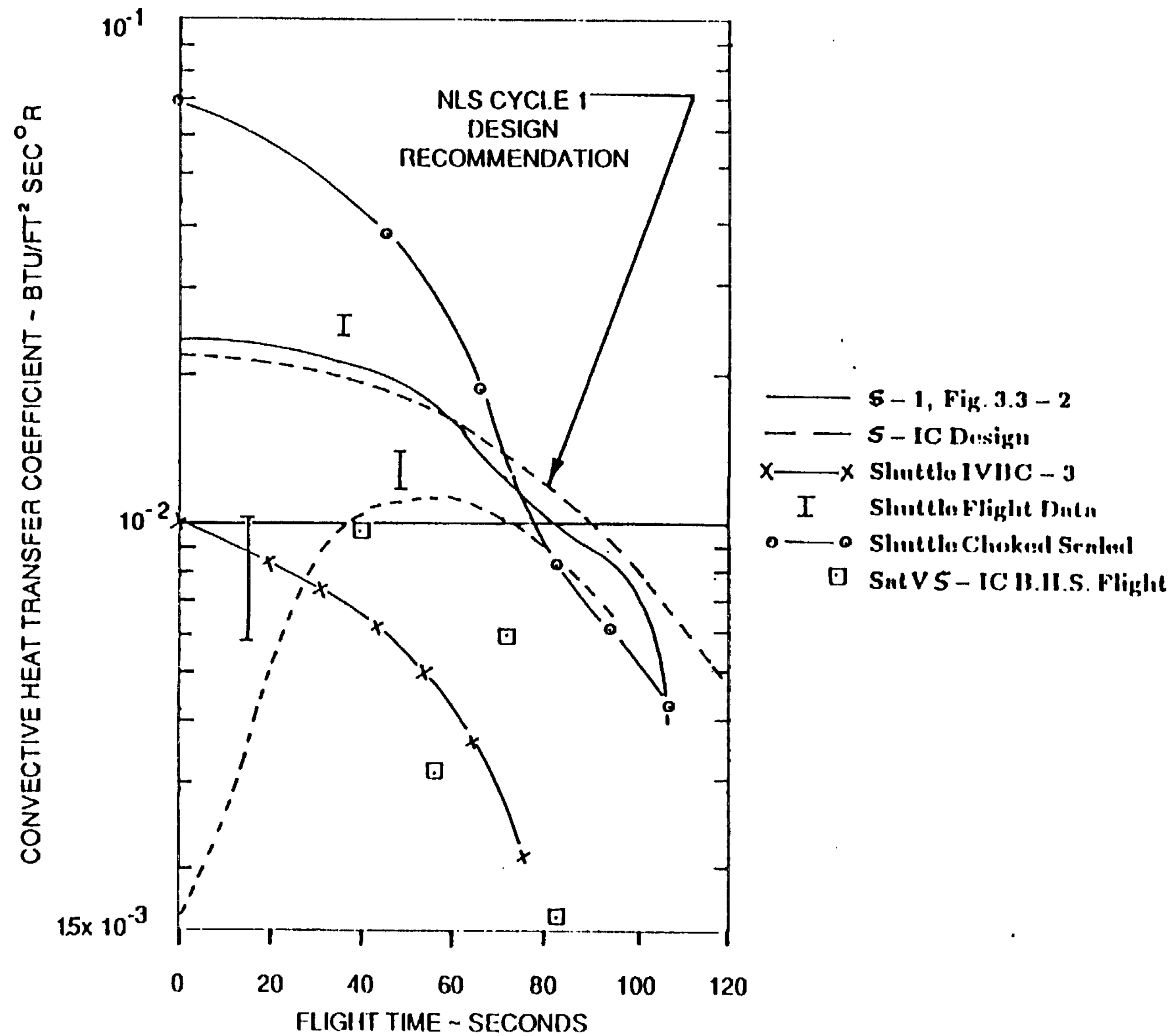


NLS - ESTIMATED CONVECTIVE BASE HEATING WITH TURBINE EXHAUST BASE BURNING

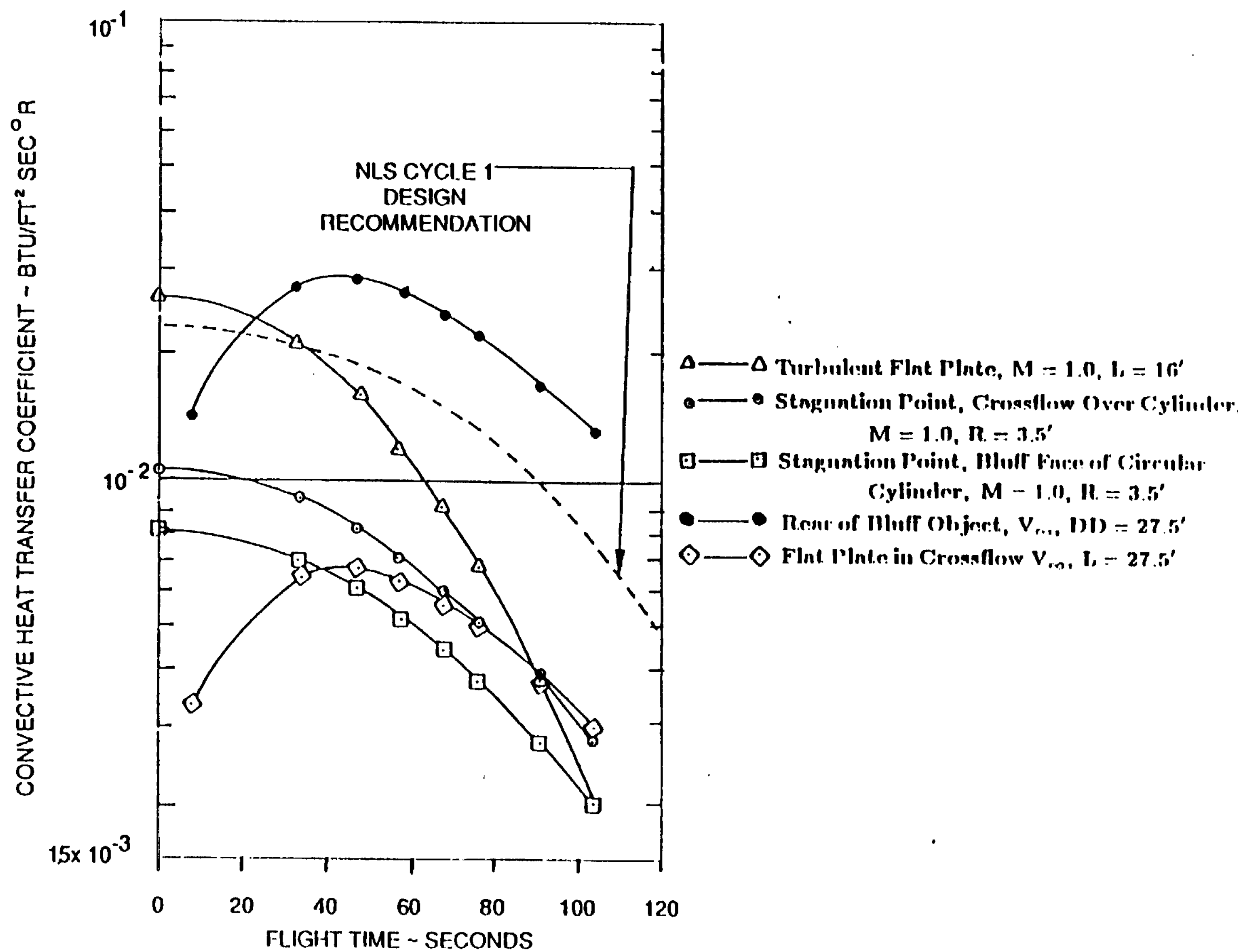
Twall = 540^0 R



NLS - CONVECTIVE HEAT TRANSFER COEFFICIENT ESTIMATES FOR CORE BASE REGION



NLS - CONVECTIVE HEAT TRANSFER COEFFICIENT ESTIMATES FOR CORE BASE REGION



RESULTS OF SHORT TERM BASE BURNING ANALYSIS

- Complex NLS base flowfields can recirculate low energy STME nozzle exhaust into base region at any altitude.
- Low energy plume boundary gases near nozzle lip will contain significant quantity of unburned H_2 and H_2O with current STME turbine exhaust disposal scheme.
- Burning of recirculate H_2 with air in base can occur from sea level to approximately 120,000 feet.
- Base gas temperatures as a result of H_2 burning approach 4000° R at low altitudes.
- Convective heat transfer coefficients on the order of 2×10^{-2} BTU/ft^2 sec$^\circ$ R are feasible in the base at typical low altitude densities and turbulence levels.
- Convective heating rates as high as 80 BTU/ft^2 sec (cold wall) are possible.

IMPLICATIONS OF PROPOSED STME DESIGN CHANGES

- Upgrading current STME design to 650K has small impact (approx. 5 to 10% increase) on Cycle 1 environments.

- If STME remains G.G. cycle engine:
  1) Variations in nozzle disposal schemes have little impact on current conservative base burning analysis approach and resulting environments.
  2) Outboard ducts change base burning potential - but have not been analyzed.

- Regenerative cooled dual combustion engine similar to SSME would effectively eliminate low altitude base burning

REMTECH

NLS

BASE HEATING/BASE BURNING
STME TURBINE EXHAUST DISPOSAL
ANALYSIS REVIEW

JANUARY 24, 1992

PREPARED BY:
ROBERT L. BENDER
REMTECH inc.
3304 WESTMILL DRIVE
HUNTSVILLE, AL 35805

BASE HEATING ENVIRONMENT COMPONENTS

REMTECH

The base heating environment is composed of a convective heating component and radiation component. Convection occurs as the base region gases flow over the base structure. Radiation to the base may be the combined radiation from several sources including: the core of the downstream plumes, the plume mixing boundaries, plume interaction regions, local hot gases in the base, localized burning in the base, or, occasionally , from other hot structures in the base. Most analysts are concerned with main plume radiation and convective heating from reversed gases.

RADIATION SOURCES

- LOW ALTITUDE (< 70 kft)
  - * Plume Core (Mach Disk)
  - * Afterburning
  - * Baseburning (Turbine Exhaust)
- HIGH ALTITUDE (> 70 kft)
  - * Plume Core (Near Field)
  - * Plume Interaction Zones
  - * Base Recirculation
- SRM SHUTDOWN SPIKE

CONVECTION SOURCES

- COOLING FROM AMBIENT AIR
- HEATING FROM RECIRCULATED PLUME GASES
  - * PLUME-PLUME INTERACTIONS
  - * PLUME-FREESTREAM INTERACTIONS
- BASE BURNING FROM RECIRCULATED TURBINE EXHAUST

HOW DOES TURBINE EXHAUST DISPOSAL AFFECT BASE HEATING?

- If turbine exhaust dumped outboard or downstream
  - Combustible gases will burn in downstream plume and are not entrained in local recirculation pattern.
  - Amount of combustible exhaust product in engine nozzle boundary layer is small — so base region convection due to recirculated gases is determined by nozzle boundary layer gas temperature.
  - Afterburning in near plume and resultant change in plume radiation is minimized.
- If turbine exhaust dumped directly in base, engine nozzle, or nozzle exit plane.
  - Local combustion of turbine exhaust gases will occur in base region when oxidizer is present and base pressure is sufficient — referred to as base burning.
  - Base burning increases base gas temperature, alters base flow patterns, and may dramatically increase base region convection and local gas radiation.
  - Nozzle injection and subsequent afterburning changes plume radiation characteristics, often increasing downstream plume radiation.

SUMMARY OF TURBINE EXHAUST DISPOSAL FLIGHT EXPERIENCE

- Flight vehicles with turbine exhaust disposal into base, engine nozzle, or external flow.
  - ATLAS
  - SATURN 1 & 1B, 1st Stage
  - SATURN V, 1st Stage
  - DELTA

  LO_2/RP-1 Propellants

  - TITAN — Aerozine 50/UDMH Propellants (Storable)
- Flight vehicles which utilized LO_2/LH_2 propellants.
  - S–IV Stage, SATURN 1
  - S–II Stage, SATURN V
  - S-IV B Stage, SATURN V

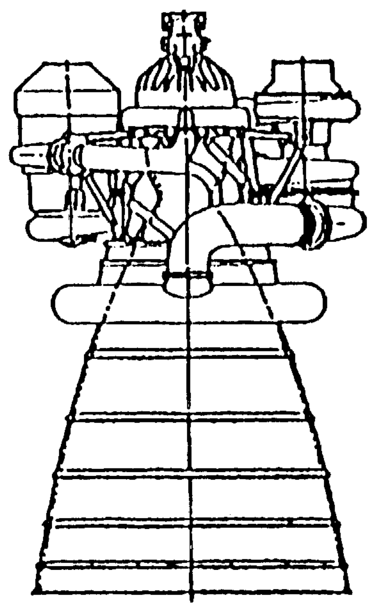
  T.E. Dumped inside nozzle-high altitude.

  - Shuttle Orbiter — Regeneratively cooled nozzle — no T.E. Discharge

THE NLS - STME TURBINE EXHAUST DILEMMA

- The STME with film/convective dump cooled nozzle:
  - is a new concept, outside experience range
  - creates potential for large mass flow of low energy, unburned H_2 at nozzle exit lip
  - H_2 will burn over wide range of mixture ratios (and pressures) with oxygen (air) present in base.
- Both NLS configurations have complex base flowfields and potential for low altitude recirculation
  - HLLV — close proximity of ASRB (with skirt) and STME (with shroud).
  - 1.5 Stage — close proximity of sustainer engines and sustainer/booster engines

STME FILM/CONVECTIVE DUMP COOLED NOZZLE



Thrust, lbs	583,000
Chamber Pressure, psia	2250
Mixture Ratio	6.0
Min. Specific Impulse (vac) sec	430.5
Weight, lbs	8000
Area Ratio	45

MAIN CHAMBER

$P_o - 2250^o\ psia$

$T_o - 6708^o\ R$

$\dot{\omega} - 1292.7\ lbm/sec$

TURBINE EXHAUST DISCHARGE

•Primary Film Coolant

$P_o - 204\ psia$

$T_o - 1190^o\ R$

$\dot{\omega} - 24.4\ lbm/sec$

•Secondary Film Coolant

$P_o - 80.3\ psia$

$T_o - 1190\ R$

$\dot{\omega} - 4.26\ lbm/sec$

•Convective Coolant

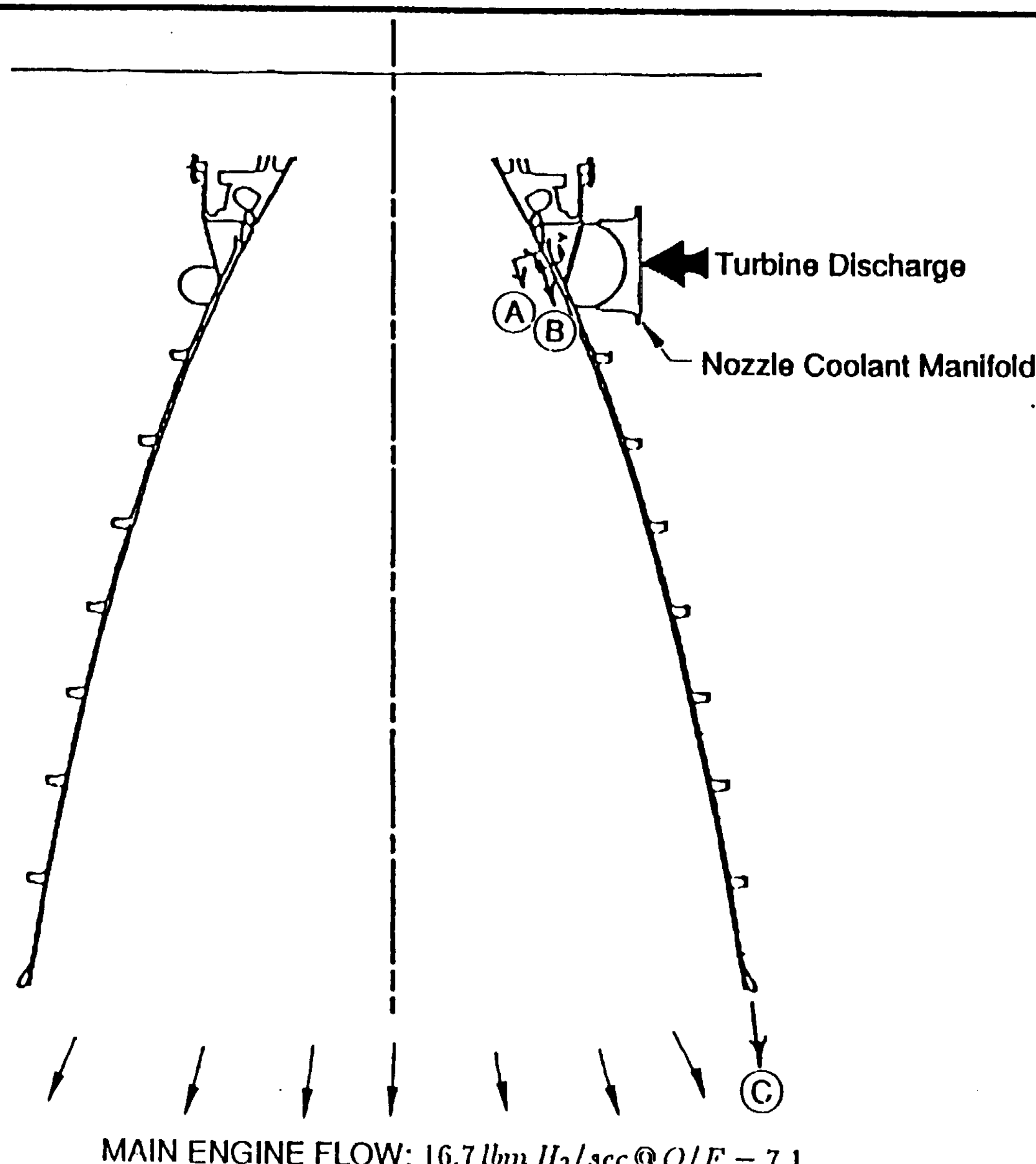
$P_o - 88.8\ psia$

$T_o - 1462.4^o\ R$

$\dot{\omega} - 35.4\ lbm/sec$

NOTE: Turbine exhaust is: 47% H_2
53% H_2O (Steam)

STME HYDROGEN FLOW RATES

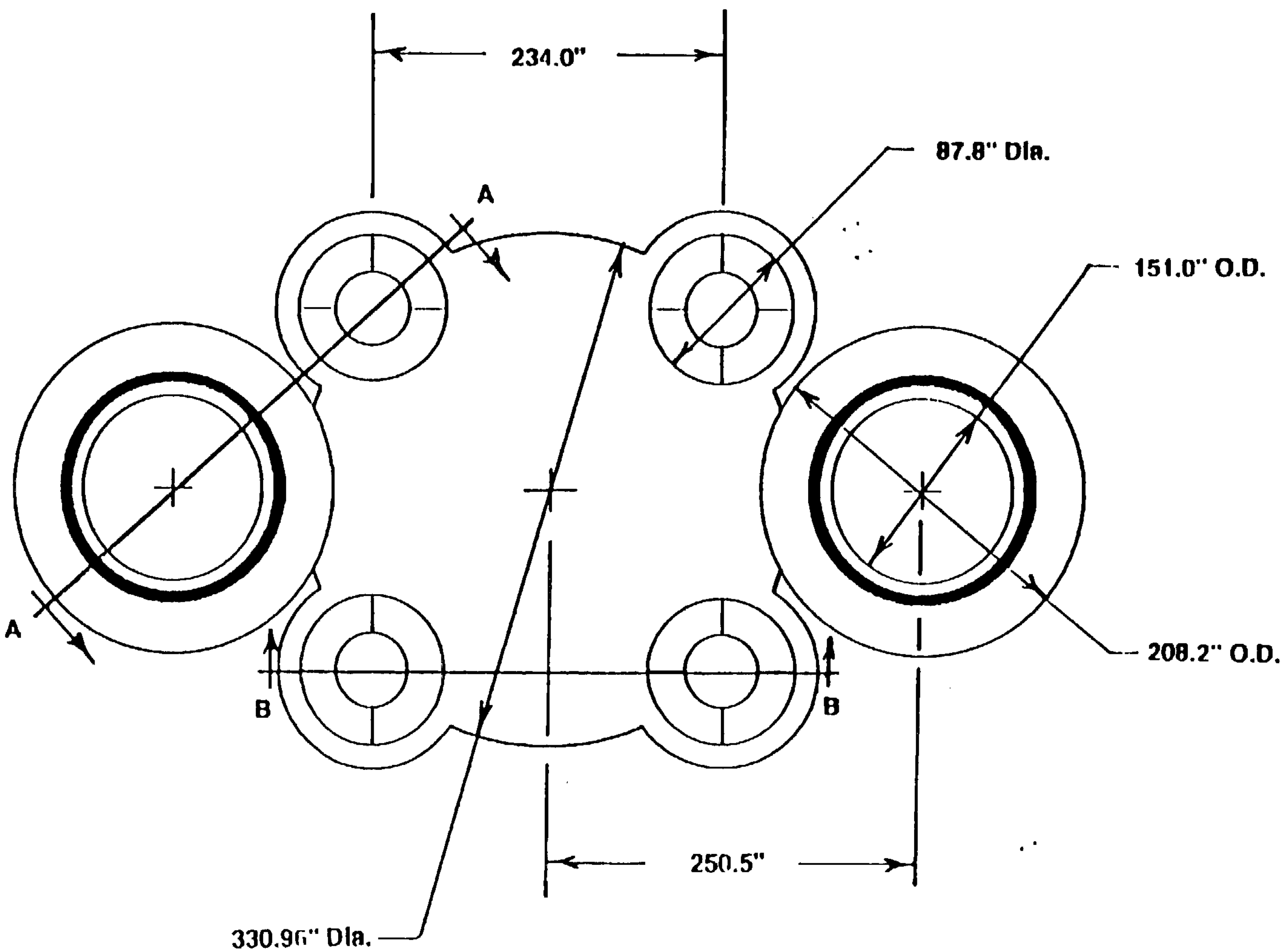


Ⓐ ***Secondary Film Coolant***
2.0 lbm H_2 /sec.

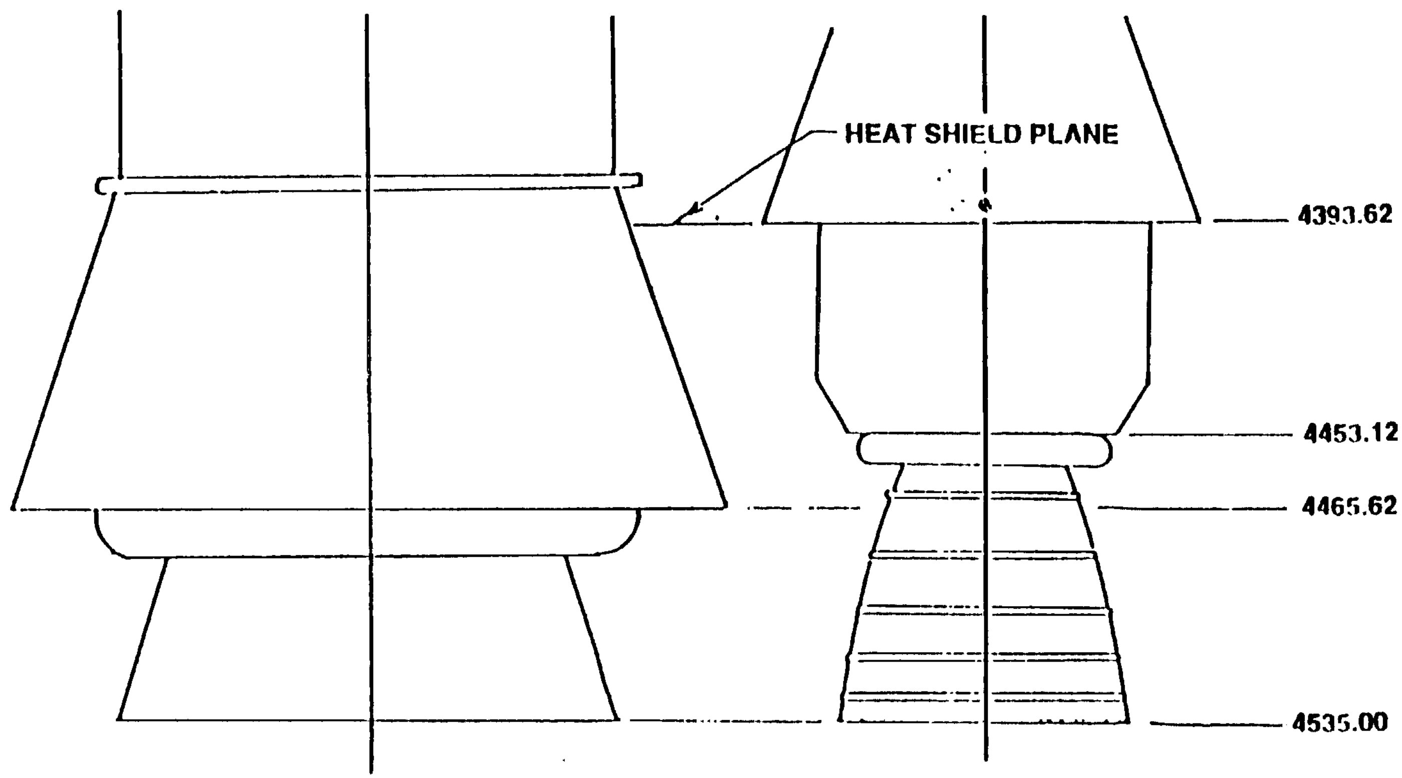
Ⓑ ***Primary Film Coolant***
11.4 lbm H_2 /sec.

Ⓒ ***Convective Dump Coolant***
16.5 lbm H_2 /sec.

IN-LINE HLLV BASE GEOMETRY

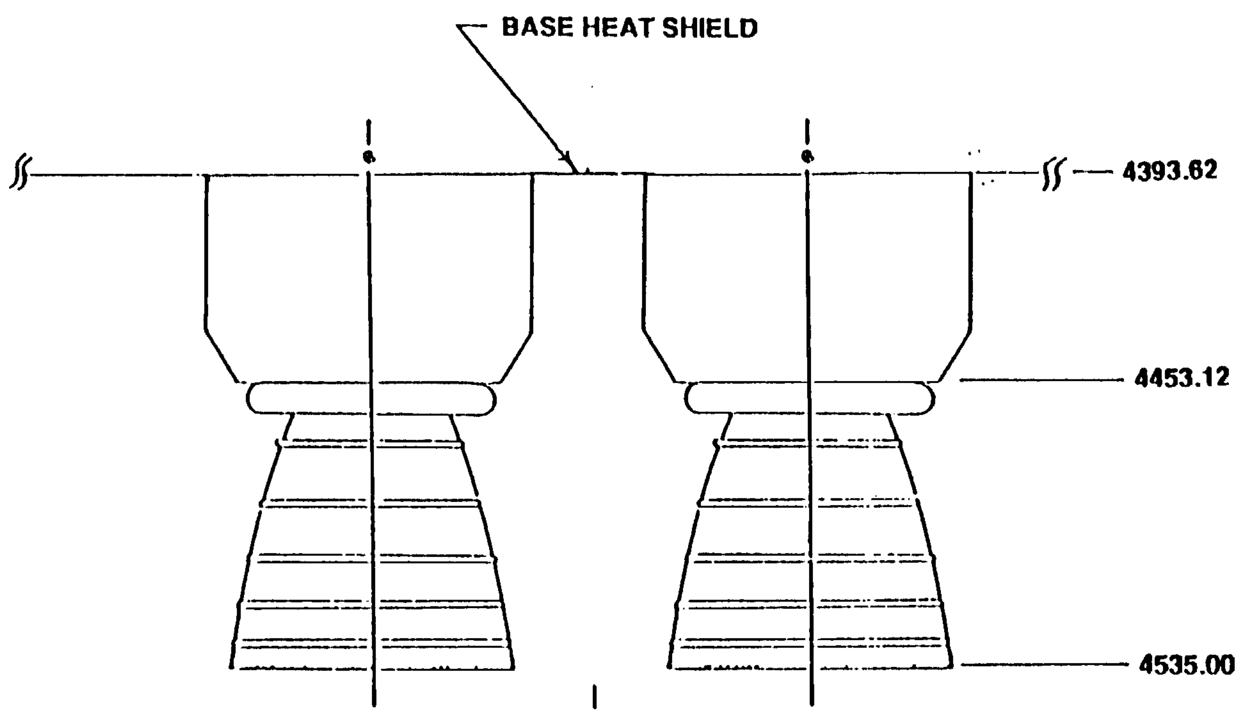


HLLV SIDE VIEW A - A



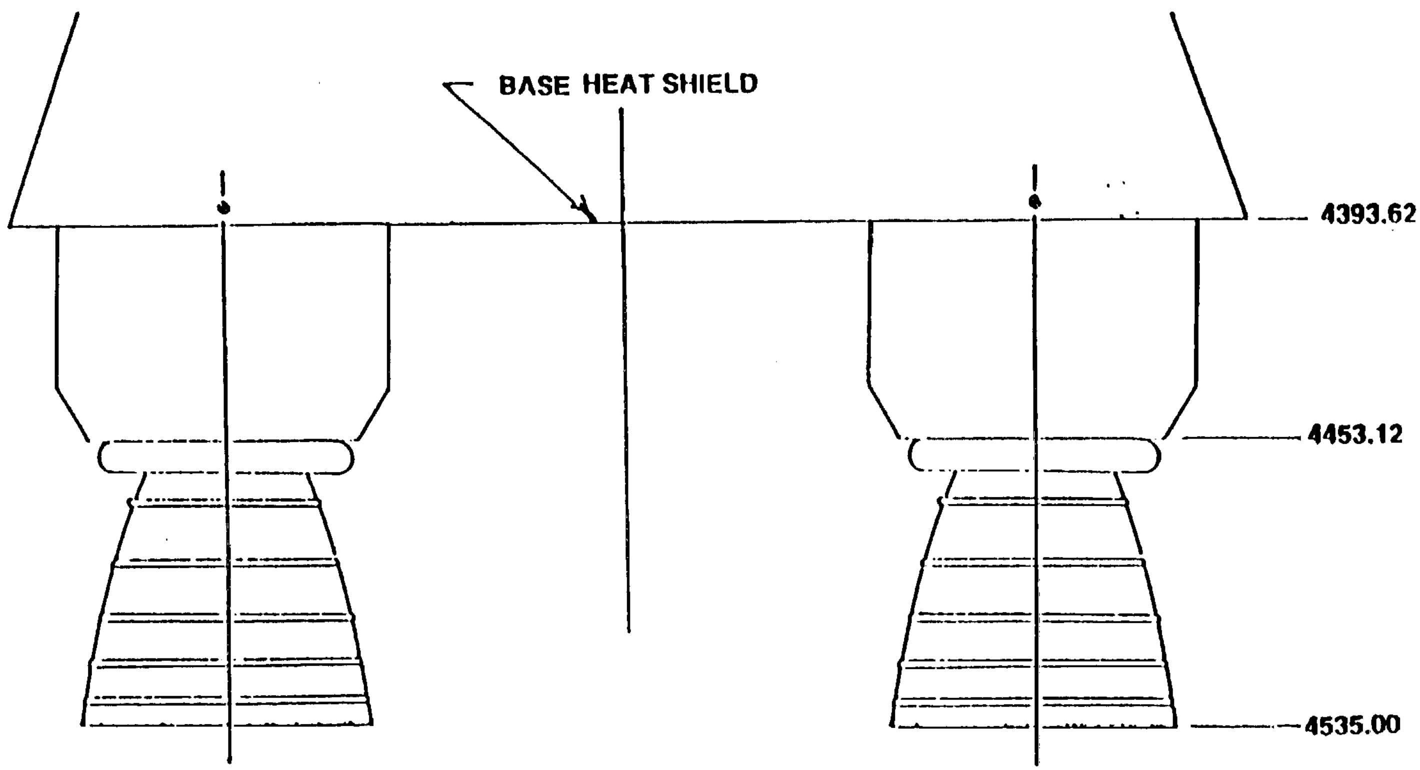
HLLV
VIEW A - A

1.5 STAGE SIDE VIEW B - B



1.5 STAGE
VIEW B - B

HLLV SIDE VIEW B - B



HLLV
VIEW B - B

NLS BASE HEATING ANALYSIS

CYCLE 1 OBJECTIVE: **Define Ascent Base Heating Environments which include**

- Latest HLLV and 1.5 Stage geometry
- Latest trajectories which maximize base heating
- Nominal plume radiation and high altitude plume recirculation convection

- PLUS -

- Radiation and convection augmentation due to base burning of STME turbine exhaust

Environments Published to Date (1/9/92)

- Preliminary Cycle 1 without base burning
  MSFC memo ED33 (98-91), Sept. 25, 1991
- Preliminary Cycle 1 with base burning
  MSFC memo ED33 (03-92), Jan. 8, 1992

Environments To Be Published

- Cycle 1 Including Updated Base Burning Analysis Results
  To be released as MSFC memo ED33 on or before January 31, 1992

NLS LOW ALTITUDE BASE BURNING ANALYSIS OBJECTIVES

$$Q_c = hc(T_{gas} - T_{wall})$$

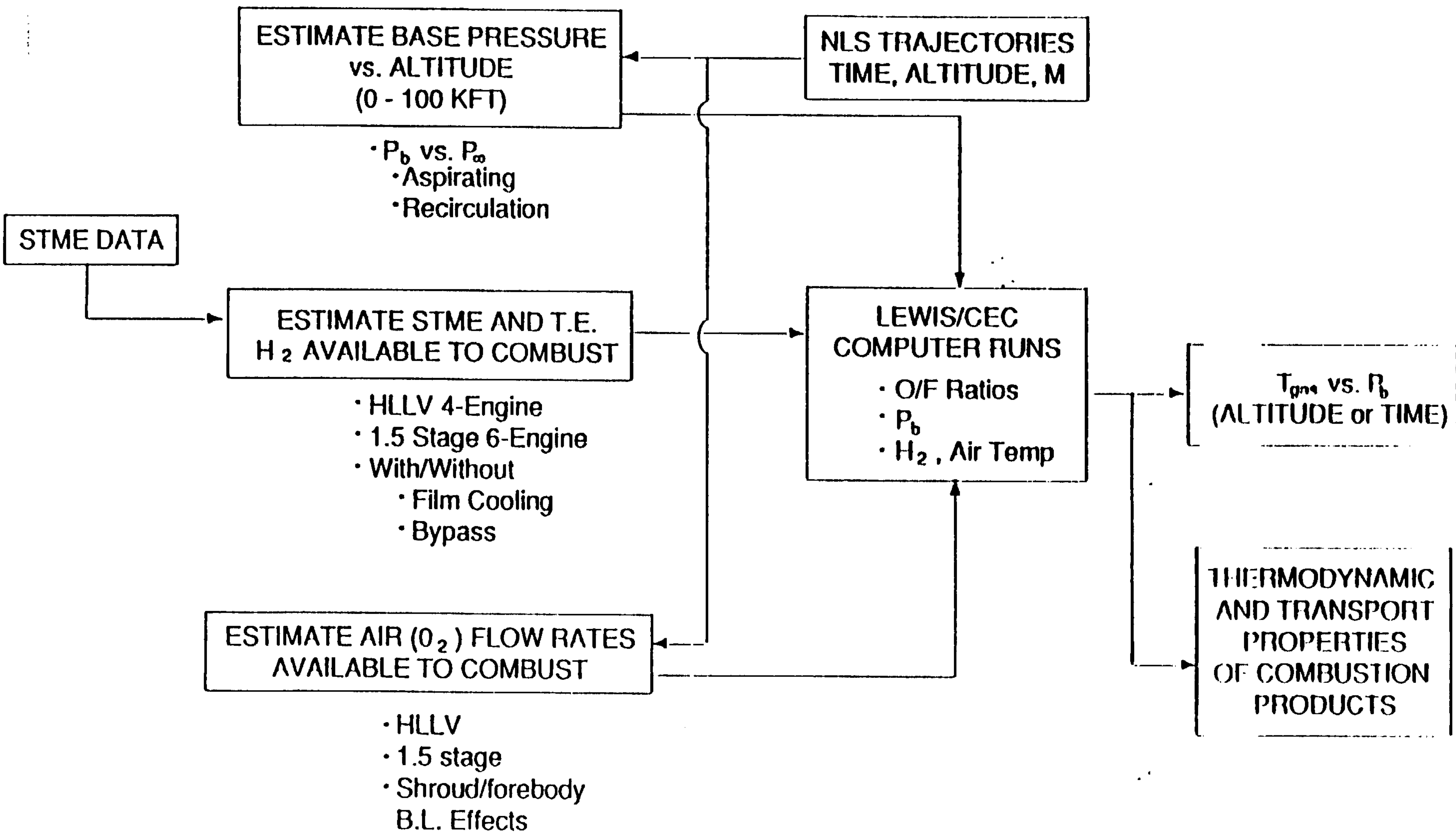
- Assuming H_2 from STME exhaust and turbine exhaust recirculated into base region and combusted with air at low altitudes.

- The analysis will:

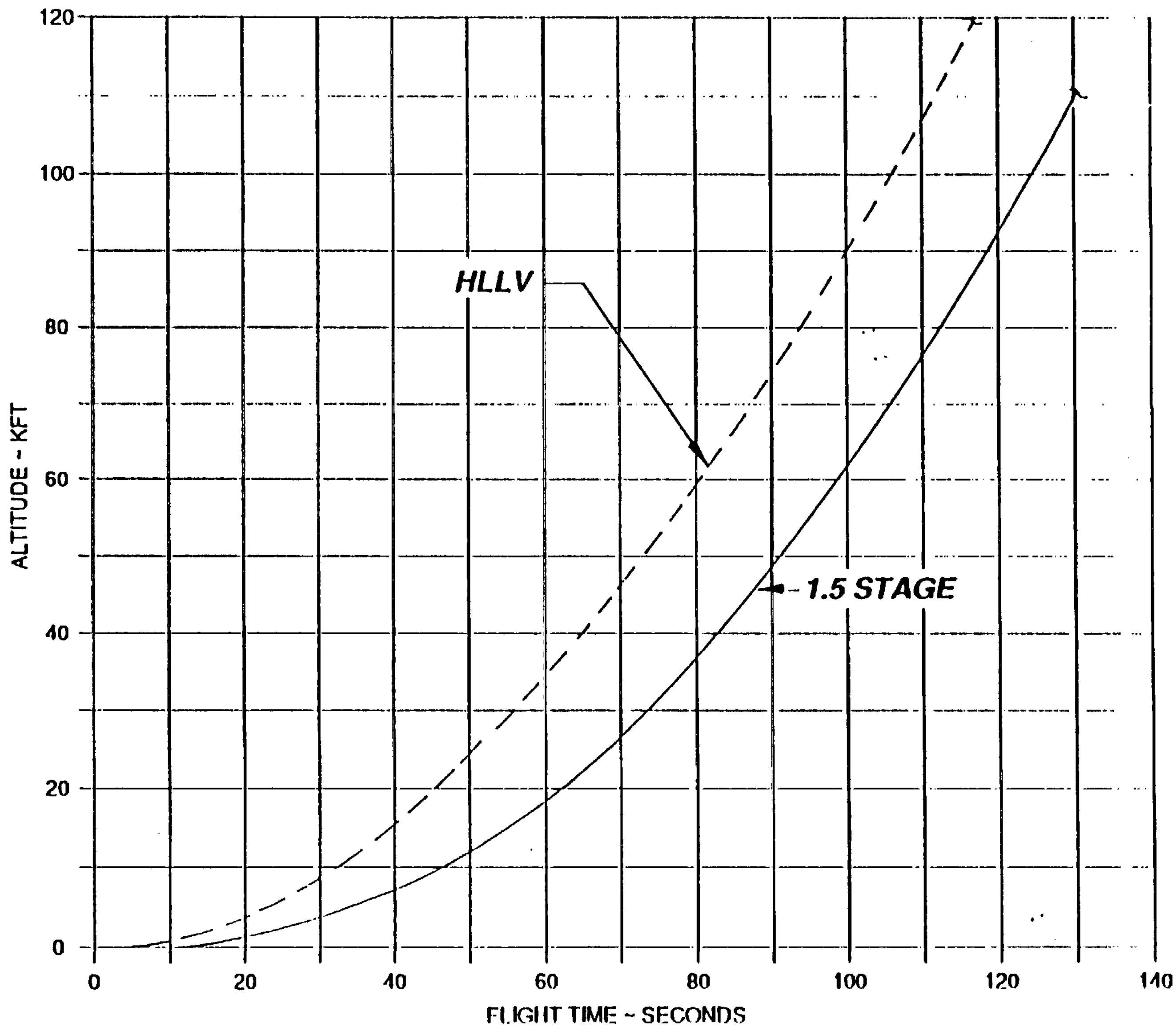
  - Define base region gas recovery temperature
  - Define convective heat transfer coefficient
  - Define upper altitude limit for H_2 air combustion
  - Compute convective heating rate

    - Base heat shield
    - STME heat shield
    - STME nozzle exterior

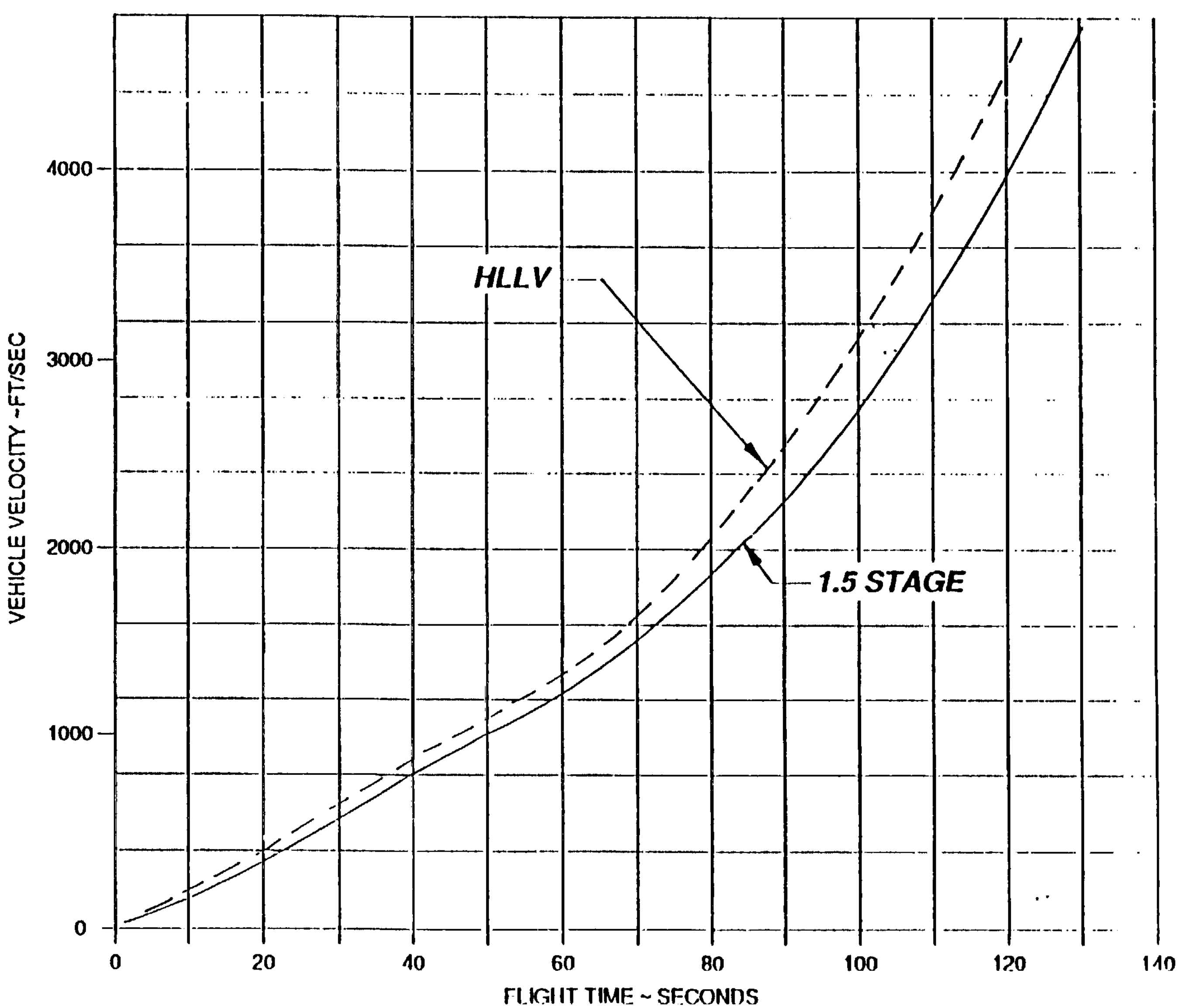
DETERMINATION OF GAS RECOVERY TEMPERATURE



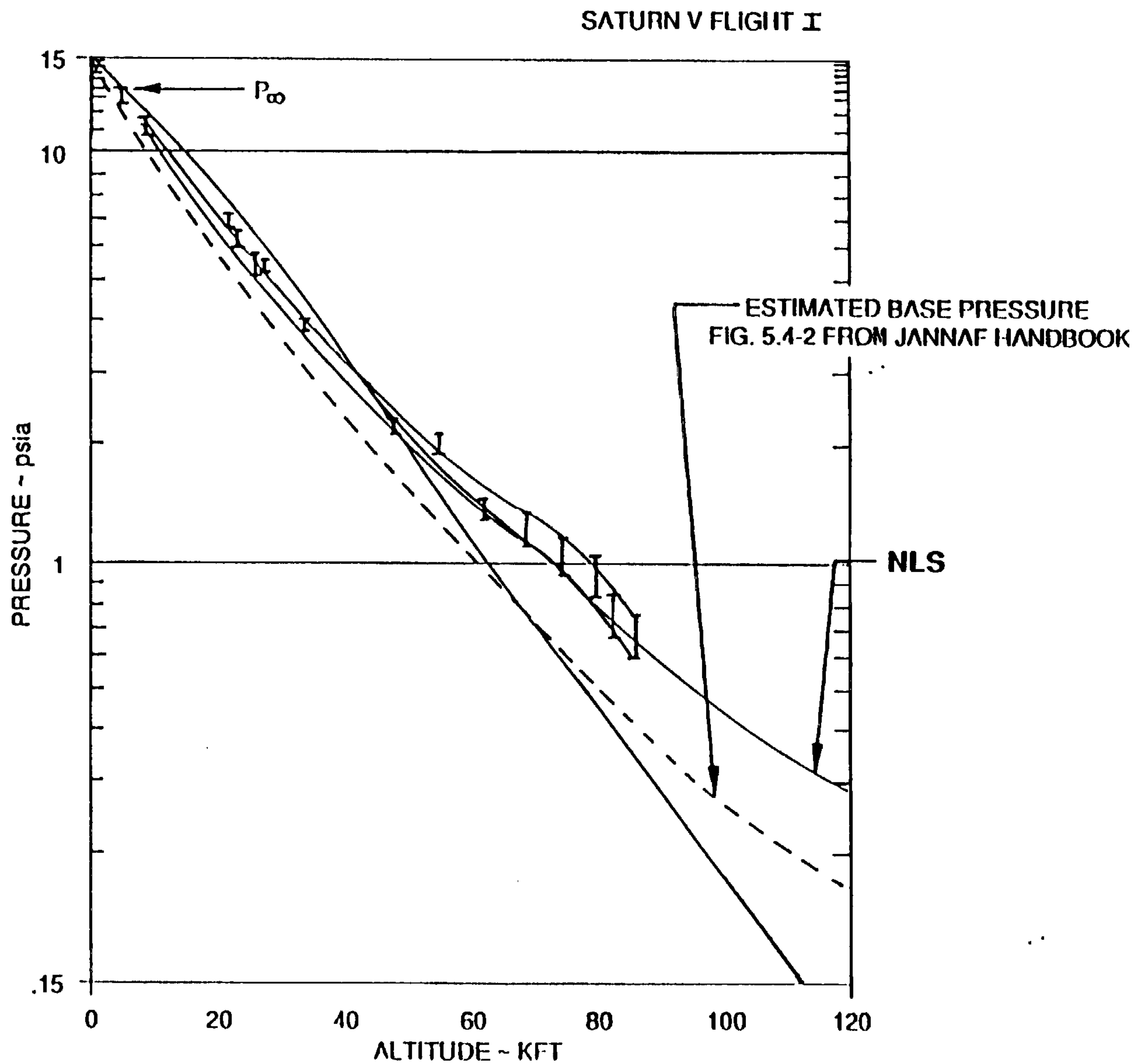
NLS BASE HEATING TRAJECTORY - ALTITUDE vs TIME



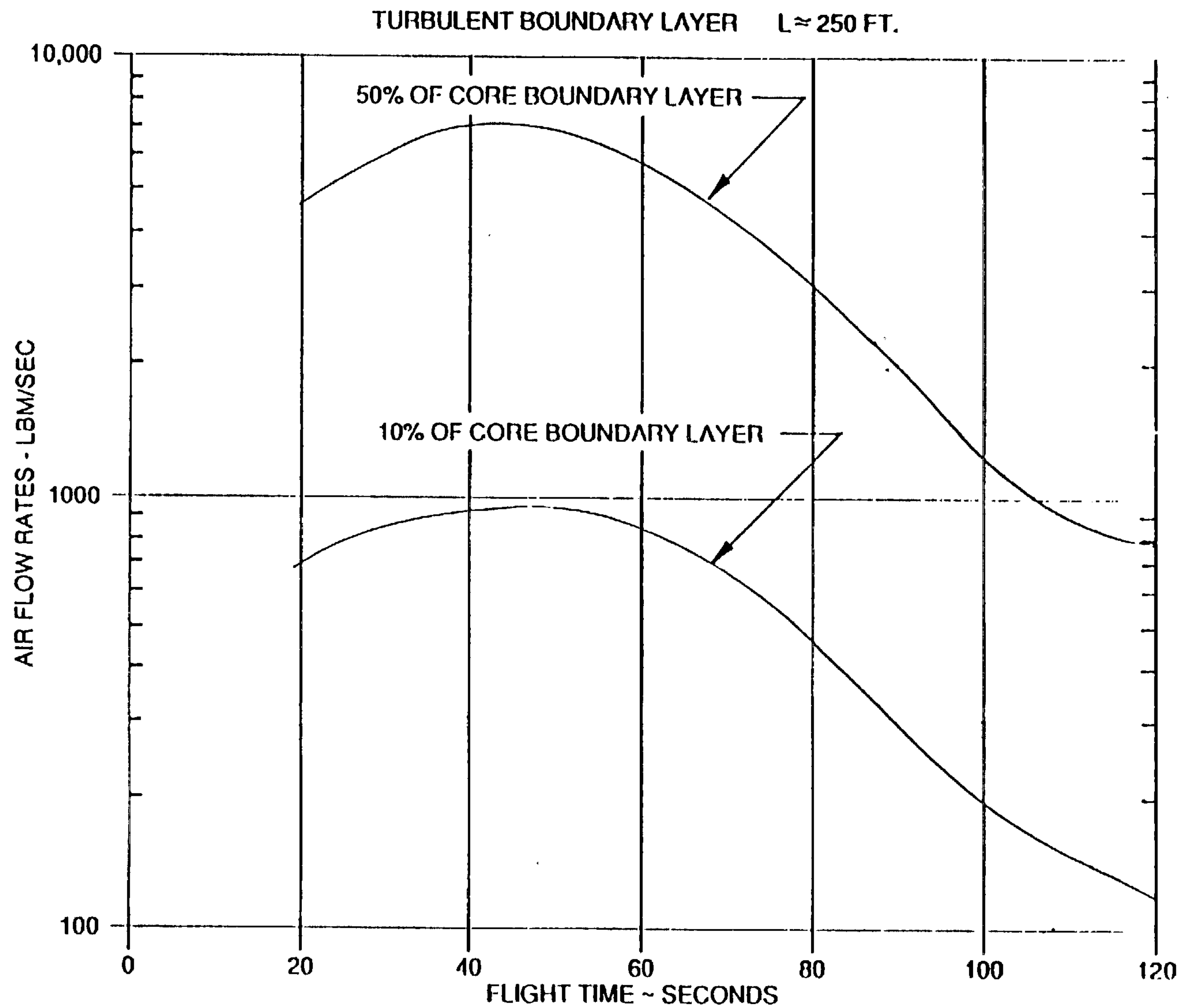
NLS BASE HEATING TRAJECTORY - VELOCITY vs TIME



NLS BASE PRESSURE ESTIMATE



NLS - ESTIMATES OF AVAILABLE AIR FLOW INTO CORE BASE REGION



NLS - ESTIMATES OF RECIRCULATED TURBINE EXHAUST INTO CORE BASE REGION

Turbine Exhaust Is: 47% H_2
53% H_2O (Steam)

HLLV — 4 STME

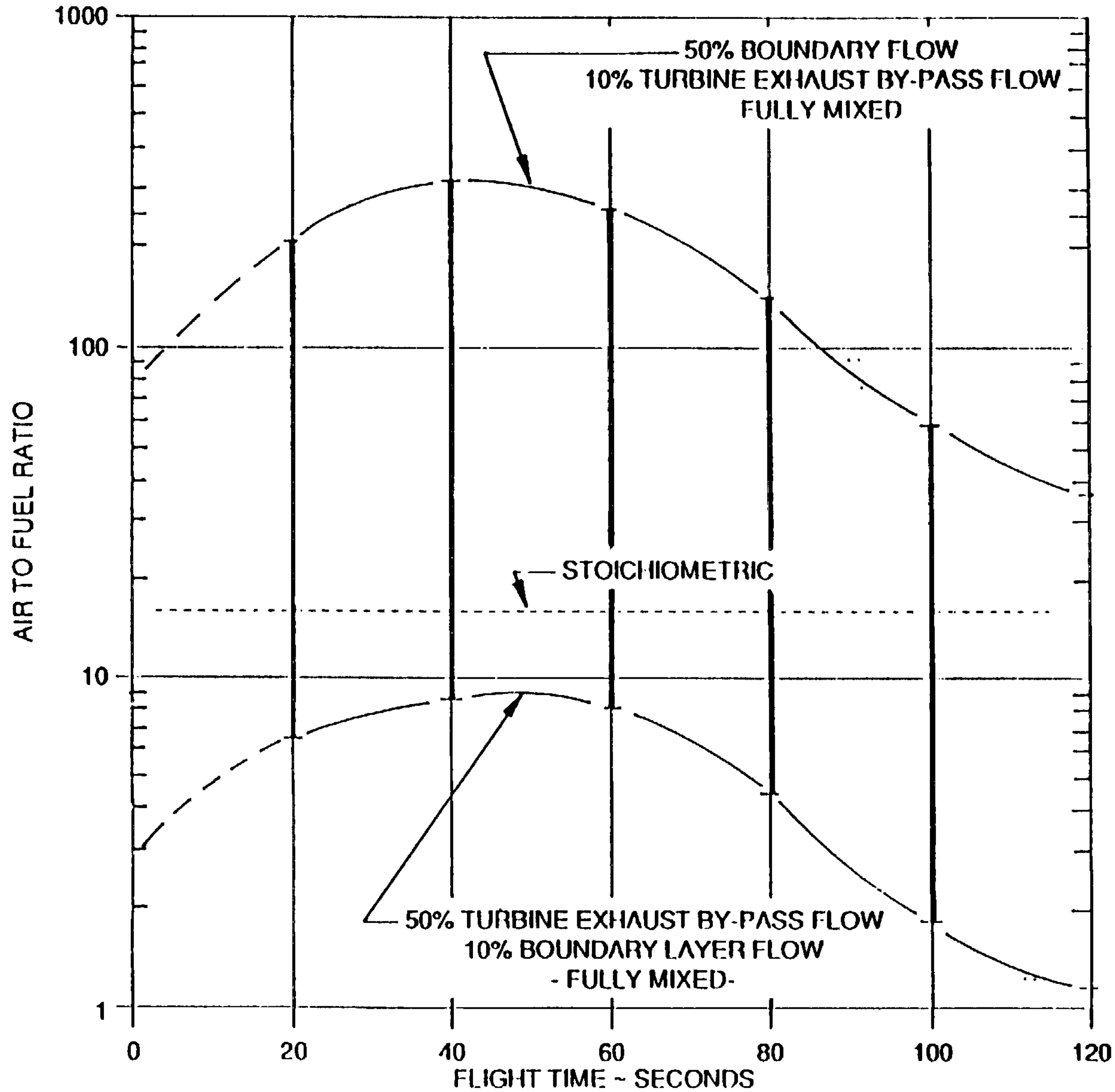
Recirculation Assumption	T.E. Recirculated lbm/sec
All Turbine Exhaust	256.2
100% Bypass	141.6
50% Bypass	70.8
10% Bypass	14.16
1% Bypass	1.416

1.5 STAGE — 6 STME

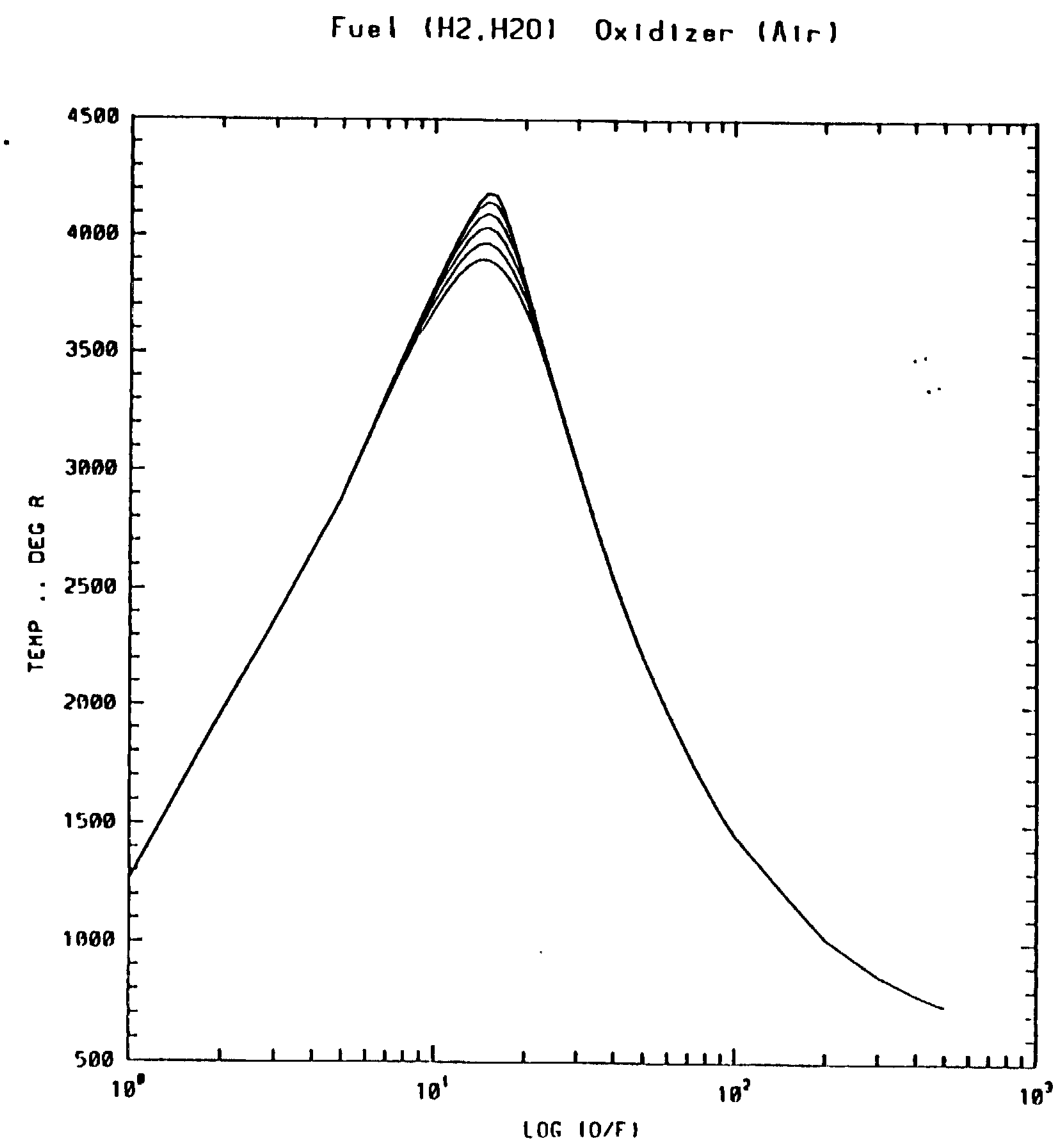
Recirculation Assumption	T.E. Recirculated lbm/sec
All Turbine Exhaust	384.36
1 Outboard STME Out	320.30
4 Outboard STME Throttled (70%)	307.49
100% Bypass	212.40
50% Bypass	106.20
10% Bypass	21.24
1% Bypass	2.124

Note: Upper limit on turbine exhaust temperature after recirculation (before mixing) is ≈ 1200°R.

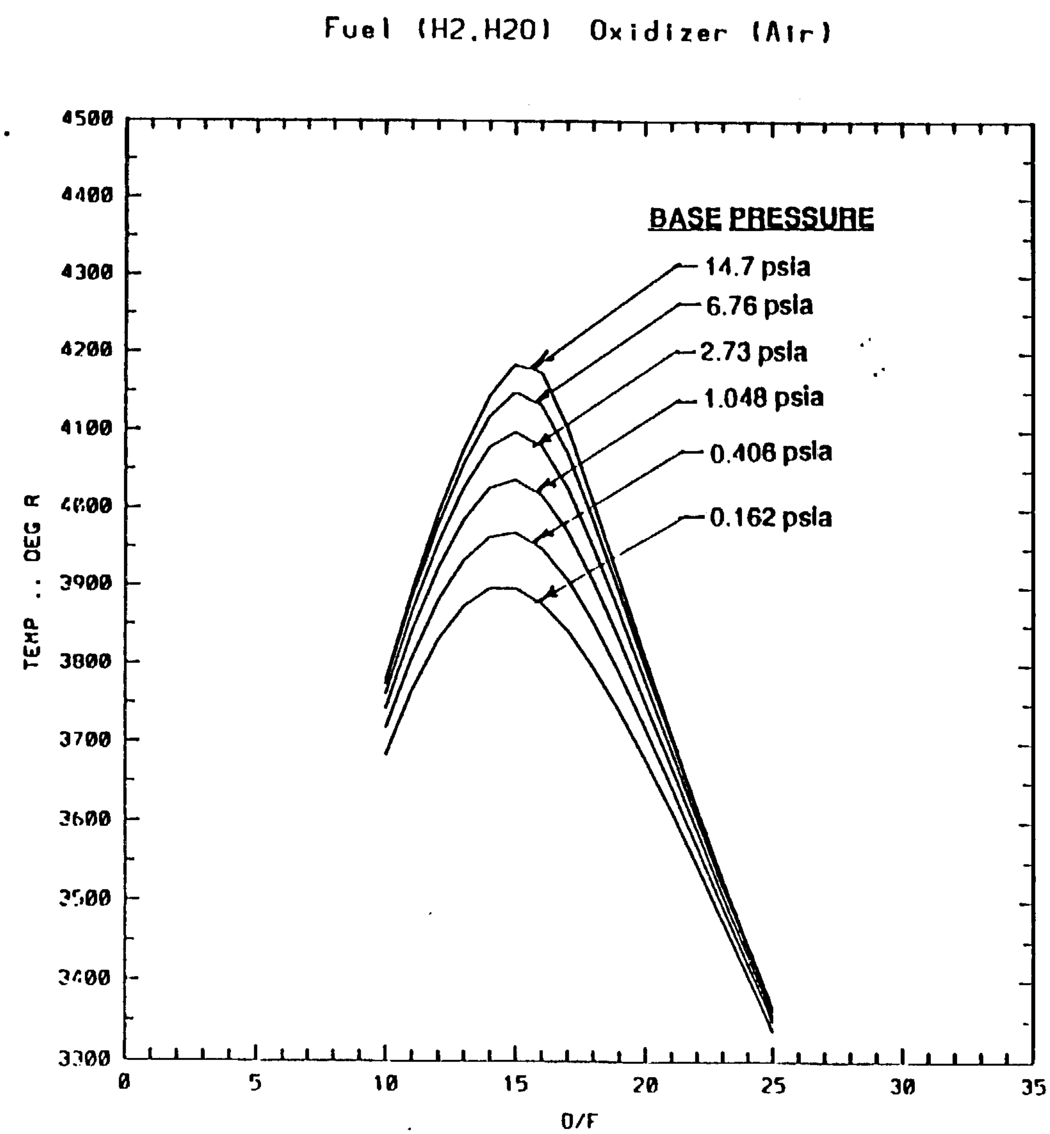
NLS - ESTIMATES OF AVAILABLE AIR FLOW INTO CORE BASE REGION



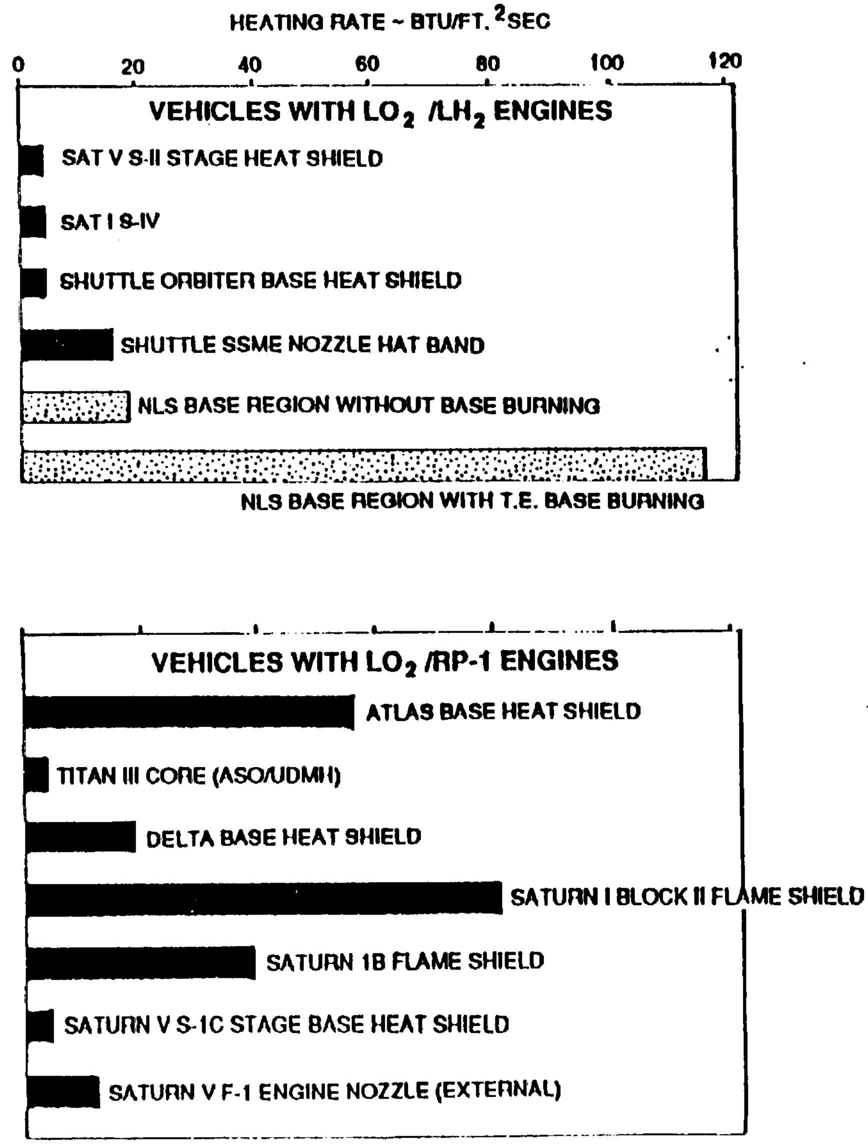
NLS - LOW ALTITUDE AIR-TURBINE EXHAUST COMBUSTION TEMPERATURES



NLS - LOW ALTITUDE AIR-TURBINE EXHAUST COMBUSTION TEMPERATURES



MAXIMUM CONVECTIVE HEATING RATES

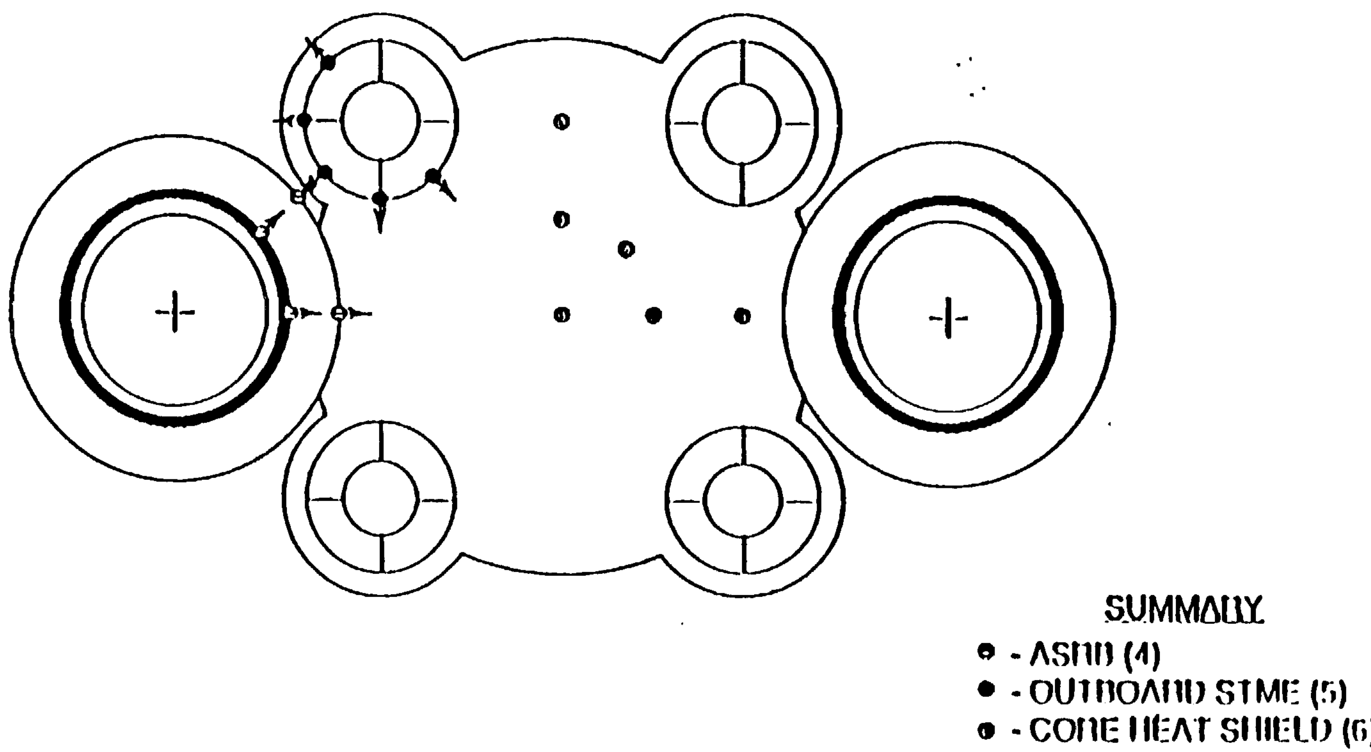


NLS BASE HEATING ENVIRONMENTS

- Without Turbine Exhaust Dumping (Assumes Regen Cooled)
  - Environment #1 — Preliminary Cycle 1 environments for nominal flight published September 25,1991
    - Reference MSFC Memo ED33 (98–81) cover to REMTECH RTN 218–03.
  - Environment #2 — Preliminary Cycle 1 environments should be increased 20% for dispersed trajectories.
- With Turbine Exhaust Dumping (Baseline STME)
  - Environment #3 — Worst case heat fluxes assuming complete combustion of turbine exhaust for nominal flight published September 25, 1991.
    - Reference MSFC Memo ED33 (98–91) cover to MSFC Memo ED31 (06–89), dated March 3, 1989
  - Environment #4 — Worst case heat fluxes should be increased by 20% for dispersed trajectories.
- Cycle 1 Environments
  - Ongoing study to refine environments through January 1992.
  - Cycle 1 environments (with and without base burning) scheduled for publication January 17, 1992.

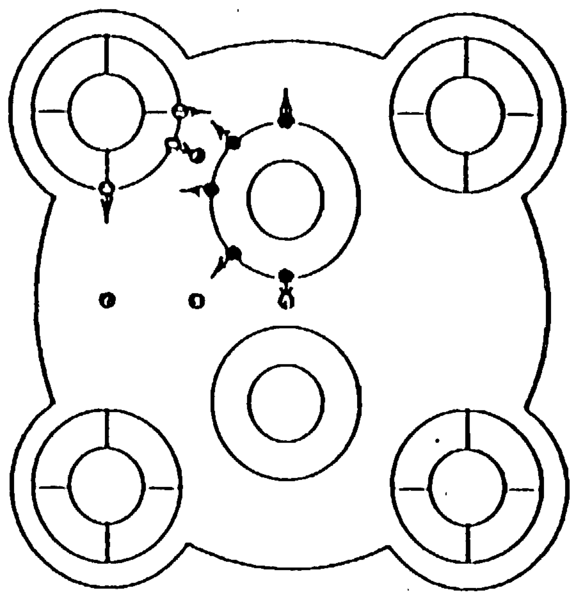
BODY POINT LOCATIONS FOR JANUARY 1992 ENVIRONMENTS

IN-LINE HLLV



BODY POINT LOCATIONS FOR JANUARY 1992 ENVIRONMENTS

1.5 STAGE REFERENCE



SUMMARY

- ⊖ - OUTBOARD STME (3)
- ● - INBOARD STME (5)
- o - CORE HEAT SHIELD (4)

NLS TURBINE EXHAUST BASE BURNING ANALYSIS PLAN

SHORT TERM (THROUGH JANUARY 1992)

- Plume Definitions
  - Estimate STME plumes with turbine exhaust dumping.
  - Develop near field plume data for sea level, 20, 40, 80, 150 kfeet altitudes.
- Radiation
  - Development new radiation plume models for various altitudes
  - Compute incident Q_{RAD} at various base locations.
- Convection
  - Estimate available H_2 in base region of various altitudes.
  - Estimate new base gas recovery temperature and heat transfer coefficient with various H_2 combustion scenarios
  - Define preliminary convective heating rate in base with H_2 burning.
- Environment
  - Replace MSFC ED31 (06–89) with updated environments for approximately 15 NLS base region body points.

NLS TURBINE EXHAUST BASE BURNING ANALYSIS PLAN

LONG TERM (AFTER JANUARY 1992)

- Continue to analyze and refine plume definitions and base flowfield thermochemistry data.
- Continue analysis of previous launch vehicle experience with various turbine exhaust disposal schemes.
- Coordinate base heating studies with STME design evolution.
- Outline test program to provide explicit thermal environment data for NLS configurations, trajectories, and STME turbine exhaust disposal schemes.
- Provide up-dated base heating environments as needed to support design evolution.

30

NLS BASE HEATING/BASE BURNING TEST PLAN

Subscale

- Cold Flow in Wind Tunnel
  - Single engine with He simulation of turbine exhaust
    - Gas Sampling
    - Base Pressure
    - Flow Visualization
  - Multiple engines with He simulation
    - Gas Sampling
    - Base Pressure
    - Flow Visualization

Note: Hot Flow in Wind Tunnel not recommended.

- Difficult
- Costly
- Not conservative due to scale effects.

NLS BASE HEATING/BASE BURNING

Conclusions/Recommendations

- Current STME T.E. Disposal scheme outside experience with previous launch vehicles.
- Current STME T.E. Disposal scheme increases potential for H_2 available to burn in base region by 120 to 1 compared with regen cooled nozzle.
- The complexity and uncertainty in base flow fields requires assumption that recirculation and burning of T.E. H_2 can occur from sea level to approximately 100,000 feet altitudes.
- Base Heating environments for Cycle 1 design should include effect of H_2 base burning and trajectory dispersions.
- Analytical studies including CFD flowfield definitions should be continued to increase understanding.
- Model test plans to simulate base recirculation should be pursued.

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BASE HEATING/BASE BURNING STME TURBINE EXHAUST DISPOSAL ENVIRONMENT REVIEW

FEBRUARY 6, 1992

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REMTECH

NLS
BASE HEATING/BASE BURNING
STME TURBINE EXHAUST DISPOSAL
ENVIRONMENT REVIEW

FEBRUARY 20, 1992

PREPARED BY:
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NLS BASE HEATING PRESENTATION OUTLINE

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- ***Background/Problem Description***
  - What is base heating?
  - How is base burning different from conventional base heating?
  - How does turbine exhaust affect base heating/base burning?

- ***Historical Review of Previous Launch Vehicles***
  - First Stage Propulsion Systems and Engine Arrangements/Base Geometry
  - Turbine Exhaust Disposal Schemes and Flight Results

- ***The NLS Base Heating/Base Burning Dilemma***
  - NLS/STME Parameters Affecting Base Heating
  - Uniqueness of the NLS Problem
  - System Constraints

- ***Chronology of NLS Base Heating Environment Development***
  - Cycle 1 Objectives
  - Schedule and Outputs

- ***Cycle 1 Base Heating Environments***
  - Radiation: Methodology and Results
  - Convection: Methodology and Results

- ***Environment Options and Near Term Implementation Plans***

BASE HEATING ENVIRONMENT COMPONENTS

REMTECH

The base heating environment is composed of a convective heating component and radiation component. Convection occurs as the base region gases flow over the base structure. Radiation to the base may be the combined radiation from several sources including: the core of the downstream plumes, the plume mixing boundaries, plume interaction regions, local hot gases in the base, localized burning in the base, or, occasionally , from other hot structures in the base. Most analysts are concerned with main plume radiation and convective heating from reversed gases.

RADIATION SOURCES

- LOW ALTITUDE (< 70 kft)
  - * Plume Core (Mach Disk)
  - * Afterburning
  - * Baseburning (Turbine Exhaust)
- HIGH ALTITUDE (> 70 kft)
  - * Plume Core (Near Field)
  - * Plume Interaction Zones
  - * Base Recirculation
- SRM SHUTDOWN SPIKE

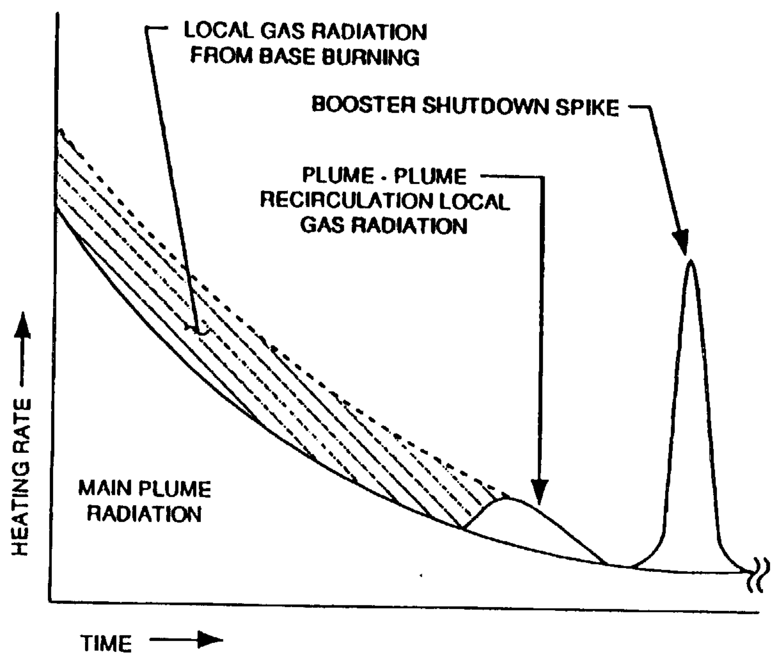
CONVECTION SOURCES

- COOLING FROM AMBIENT AIR
- HEATING FROM RECIRCULATED PLUME GASES
  - * PLUME-PLUME INTERACTIONS
  - * PLUME-FREESTREAM INTERACTIONS
- BASE BURNING FROM RECIRCULATED TURBINE EXHAUST

BASE BURNING vs CONVENTIONAL BASE HEATING

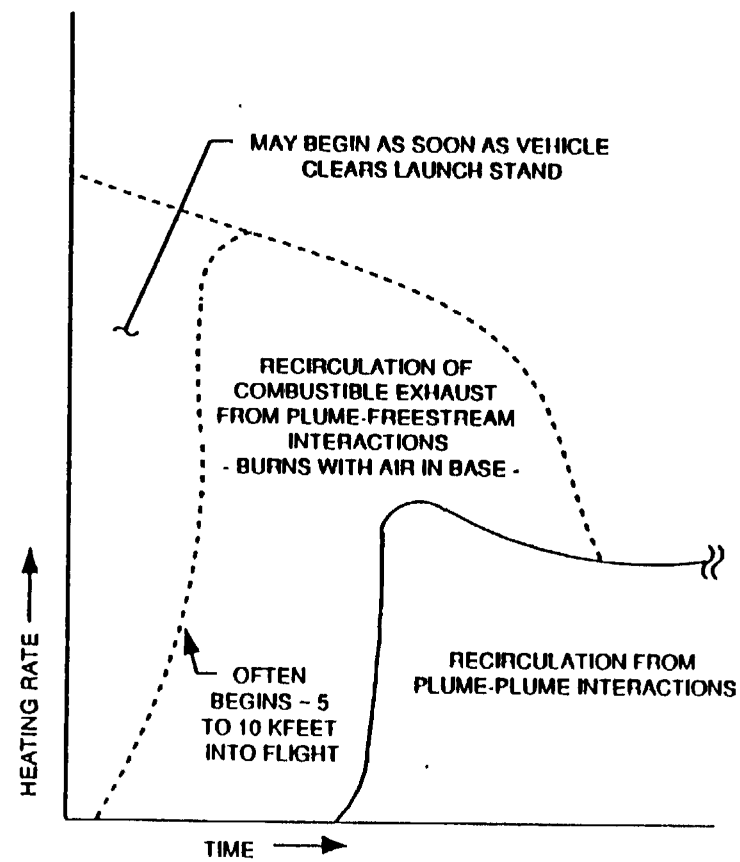
RADIATION

- Base burning increase in radiation normally small compared with conventional radiation



CONVECTION

- Base burning convection may be large in relation to conventional convection



SUMMARY OF TURBINE EXHAUST DISPOSAL FLIGHT EXPERIENCE

- Flight vehicles with turbine exhaust disposal into base, engine nozzle, or external flow.
  - ATLAS
  - SATURN 1 & 1B, 1st Stage
  - SATURN V, 1st Stage
  - DELTA

  (ATLAS through DELTA:) LO_2/RP-1 Propellants

  - TITAN — Aerozine 50/UDMH Propellants (Storable)
- Flight vehicles which utilized LO_2/LH_2 propellants.
  - S–IV Stage, SATURN 1
  - S–II Stage, SATURN V
  - S-IV B Stage, SATURN V

  (S–IV through S-IV B:) T.E. Dumped inside nozzle-high altitude.

  - Shuttle Orbiter — Regeneratively cooled nozzle — no T.E. Discharge

PAST EXPERIENCE WITH TURBINE EXHAUST DISPOSAL
--- LARGE U.S. LAUNCH VEHICLES ---

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VEHICLE	T.E. DISPOSAL SCHEME	EXPERIENCE/LESSON LEARNED
JUPITER -1A	• Duct Along Nozzle to Exit Plane • Change to Outboard Duct	• 1st Flight Failed Due to Base Heating • No failure
ATLAS	• Duct into Base - By Center Engine • Change to Outboard Duct	• 1st 2 Flights Failed Due to Base Heating • No Failure
DELTA	• Duct through Heat Shield	• High local heating on heat shield while SRM's attached
TITAN II	• Two ducts exiting slightly aft of boattail base. • Strong air scooping eliminates base burning.	• Heating not severe • No failure due to T.E. burning
TITAN III (Core)	• Core engine ignited at $H \geq 100$ kft; above altitude of serious burning.	• No trouble
SATURN I	• Inbd engine ducted to fin outbd of base • Outbd engine into nozzle through exhausterator.	• High heating early in flight • No failure due to T.E. burning
SATURN IB	• Inbd engine ducted through 4 crescent opening in flame shield • Exhausterator on outbd engine	• T.E. exhaust did not burn; cooled flame shield • No failure
SATURN V	S-IC Stage — F-1 Engine T.E. Dumped in Nozzle @ $A/A^{*}=10$	• No Failure Due to Base Heating • Unburned RP-1 Afterburning in Plume @ Low Altitude, Burned in Base @ High Altitude
NSTS SPACE SHUTTLE	• No T.E. Disposal on SSME • SRB T.E. Dumped Outboard	• No Failure Due to Base Heating • Predictable Environments

THE NLS - STME TURBINE EXHAUST DILEMMA

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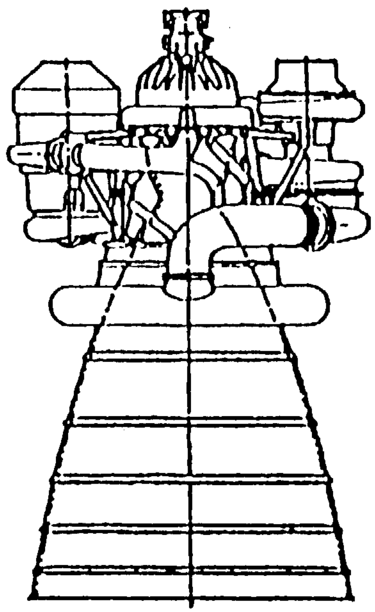
- The STME with film/convective dump cooled nozzle:
  - is a new concept, outside experience range
  - creates potential for large mass flow of low energy, unburned H_2 at nozzle exit lip
  - H_2 will burn over wide range of mixture ratios (and pressures) with oxygen (air) present in base.
- Both NLS configurations have complex base flowfields and potential for low altitude recirculation
  - <u>HLLV</u> — close proximity of ASRB (with skirt) and STME (with shroud).
  - <u>1.5 Stage</u> — close proximity of sustainer engines and sustainer/booster engines

STME FILM/CONVECTIVE DUMP COOLED NOZZLE



Thrust, lbs	683,000
Chamber Pressure, psia	2250
Mixture Ratio	6.0
Min. Specific Impulse (vac) sec	430.6
Weight, lbs	8000
Area Ratio	45

MAIN CHAMBER

$P_o - 2250^\circ\ psia$

$T_o - 6708^\circ\ R$

$\dot{\omega} - 1292.7\ lbm/sec$

TURBINE EXHAUST DISCHARGE

•Primary Film Coolant

$P_o - 204\ psia$

$T_o - 1190^\circ\ R$

$\dot{\omega} - 24.4\ lbm/sec$

•Secondary Film Coolant

$P_o - 80.3\ psia$

$T_o - 1190\ R$

$\dot{\omega} - 4.26\ lbm/sec$

•Convective Coolant

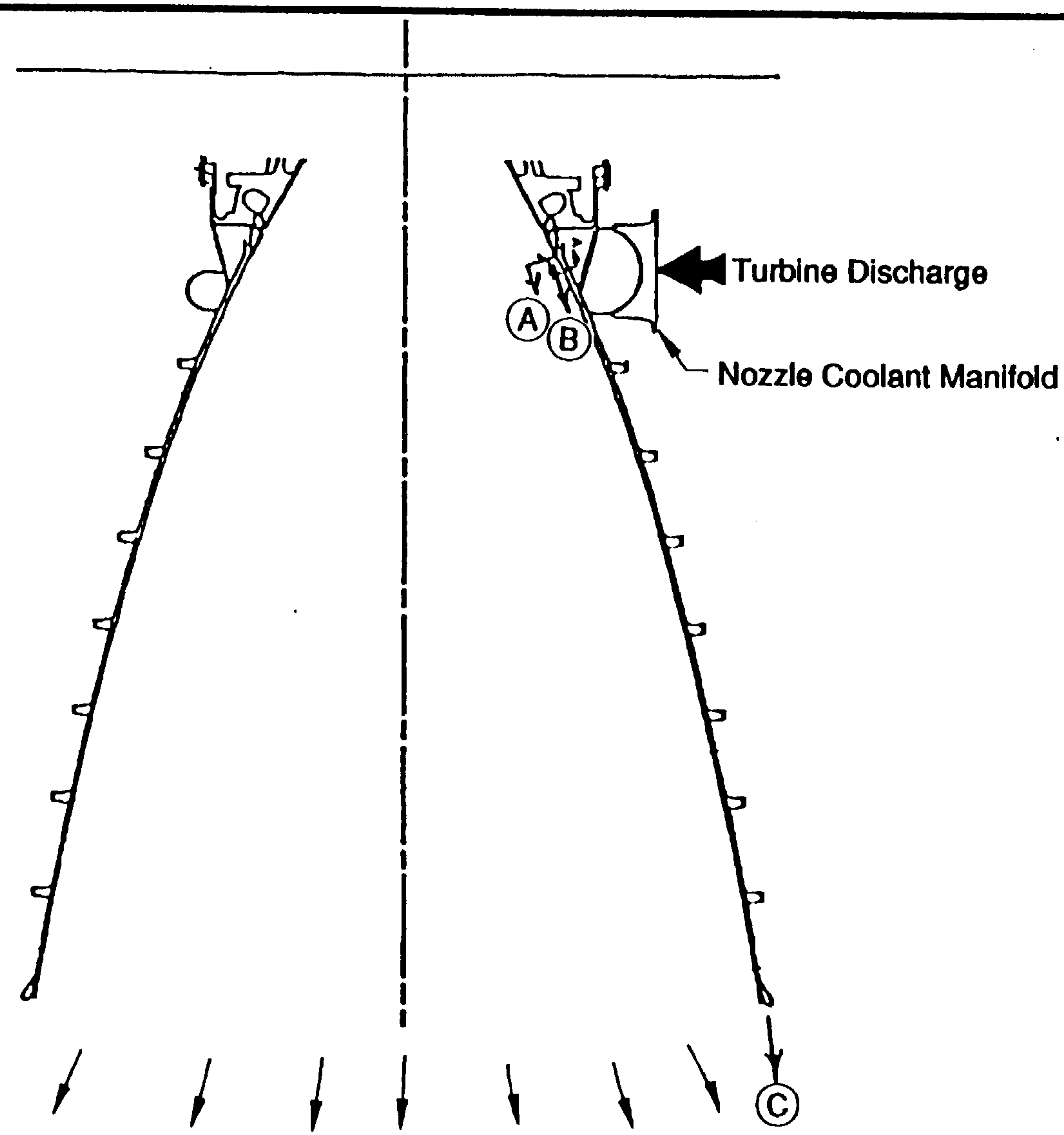
$P_o - 88.8\ psia$

$T_o - 1462.4^\circ\ R$

$\dot{\omega} - 35.4\ lbm/sec$

NOTE: Turbine exhaust is: 47% H_2
53% H_2O (Steam)

STME HYDROGEN FLOW RATES



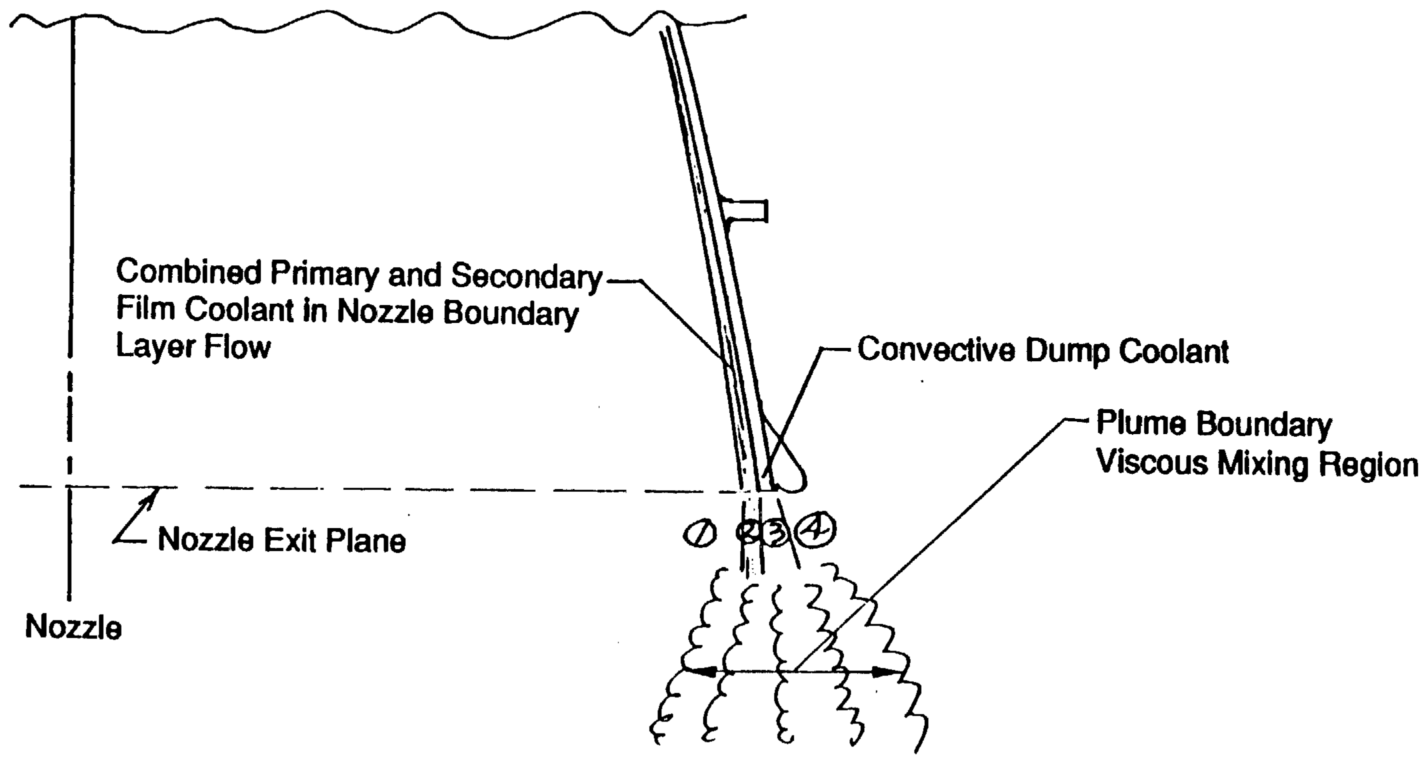
(A) ***Secondary Film Coolant***
2.0 lbm H_2/sec.

(B) ***Primary Film Coolant***
11.4 lbm H_2/sec.

(C) ***Convective Dump Coolant***
16.5 lbm H_2/sec.

MAIN ENGINE FLOW: $16.7\, lbm\, H_2/sec$ @ $O/F = 7.1$

STME PLUME EXPANSION/RECIRCULATION FLOWFIELD



<u>Four (4) Stream Mixing Problem</u>

1) Nozzle Inviscid Flow - $P_o \approx 2200$ psia
2) Film Coolant/Nozzle Boundary Layer Flow - $P_o \approx 200$ psia
3) Convective Dump Coolant Flow - $P_o \approx 90$ psia
4) Freestream or Base Region Flow - $P_o \approx 14.7$ or less psia

F-1 ENGINE/STME COMPARISONS

Comparison Parameter	F-1 Engine	STME
Operating Conditions		
• Chamber Pressure, PSI	1126/983	2250
• Chamber Temperature (°R)	6383	6708
• Area ratio	16	45
• Propellants	LOX/RP-1	LO_2/LH_2
• O/F	2.27	7.1
• 1D Exit Pressure, PSIA	6.18	
Nozzle Description		
• Exit Diameter	140″	87.8″
• Nozzle Half Angle	13°	
Flow Rates		
Main Chamber, lbm/sec	5564.4	1292.7
Turbine Exhaust Total lbm/sec	170.5	64.06
O/F	0.42	
Turbine Exhaust, Fuel Only, lbm/sec	120.3 RP-1	29.9 H_2

F-1 ENGINE/STME COMPARISONS

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Comparison Parameter	F-1 Engine	STME
Ratios		
1. $\frac{\text{Total Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0306	0.0496
2. $\frac{\text{Combustible Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0216	0.0231
T. E. Characteristics		
• Total Pressure, PSIA	57	204/89
• Temperature, °	1465°F	1190/1462°R

SATURN V/S-1C STAGE/NLS 1.5 STAGE

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COMPARISON PARAMETER	S-1C STAGE F-1 ENGINE	1.5 STAGE STME	COMMENT
Base Geometry			
1. Base Diameter, ~ Inches	396"	330.96"	
2. Length from Stage Center to Outboard Engine, ~ Inches	182"	165.5	
3. Length from Base Heat Shield to Nozzle Exit Plume, ~ Inches	227.4"	141.4"	
4. Nozzle Exit Diameter, ~ Inches	140"	87.8"	
5. Overall Vehicle Length, ~ Inches	4416	3221.3	
6. Shroud Length Below Base Heat Shield, Inches (Overhang)	63.5"	0.0	
7. Shroud Angle, ~ Degrees	15	20	
8. Outboard Shroud Height, Inches	77"	63"	
RATIOS			
9. $\frac{\text{Total Eng. Exit Area}}{\text{Base Area}}$	0.6249	0.4223	1.5 Stage Base More Open
10. $\frac{\text{Engine Length}}{\text{Base Diameter}}$	0.5742	0.4263	S-1C Engines Extend Further Aft
11. $\frac{\text{Nozzle Exit Diameter}}{\text{Engine Length}}$	0.62	0.6209	Same
12. $\frac{\text{Center to Outboard ℄ Distance}}{\text{Nozzle Exit Diameter}}$	1.3	$\boxed{1.885}$ $\frac{10.86'}{7.32'} = 1.48$ $\frac{9.92'}{7.32'} = 1.35$	Center to Outboard Larger on 1.5 Stage Center to Center ≈ same
13. $\frac{\text{Forebody Length}}{\text{Base Diameter}}$	11.15	9.73	S-1C (Saturn V) More Slender

WHY IS NLS/STME BASE BURNING PROBLEM UNIQUE?

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- Although general flow patterns similar to Saturn V S-1C Stage, shroud and booster geometry, number of STMEs and STME length create ***unique*** base flow field for NLS

- Current STME disposal scheme creates 4 stream mixing problem at nozzle lip which is ***unique*** and different from H-1 and F-1 engines exhausterator and manifold/slot injection schemes

- H_2 injection pressures on STME higher than H-1 or F-1 which may enhance diffusion into main plume flow but also changes momentum and turbulence in shear mixing layer - creating unique recirculation potential

- H_2 potential for burning and high energy release from combustion uniquely different from RP-1 (Kerosene)
  - H_2 has wider combustion limit than RP-1
  - H_2 has 3 to 5 times energy release of RP-1 per lb.

  NOTE: RP-1 loses energy in soot formation

- Stoichiometric burning temperatures of H_2 slightly higher than RP-1 when burned with air at comparable pressure

- Transport properties of H_2 /air combustion products different from RP-1/air products; results in different convective heating over comparable surfaces

NLS BASE HEATING ANALYSIS

CYCLE 1 OBJECTIVE: Define Ascent Base Heating Environments which include

- Latest HLLV and 1.5 Stage geometry
- Latest trajectories which maximize base heating
- Nominal plume radiation and high altitude plume recirculation convection

- PLUS -

- Radiation and convection augmentation due to base burning of STME turbine exhaust

Environments Published to Date (1/9/92)

- Preliminary Cycle 1 without base burning
  MSFC memo ED33 (98-91), Sept. 25, 1991
- Preliminary Cycle 1 with base burning
  MSFC memo ED33 (03-92), Jan. 8, 1992

Environments To Be Published

- Cycle 1 Including Updated Base Burning Analysis Results
  MSFC memo ED33 (15-92), Feb. 7, 1992

NLS CYCLE 1 BASE HEATING METHODOLOGY

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PRELIMINARY CYCLE 1 METHODOLOGY

$$Q_{Total} = Q_{Rad} + Q_{Conv}$$

- ***RADIATION***
  - ASRM:
    - Viewfactor predictions using Cycle 1 sea-level plume model
    - Modified Cycle 1 altitude adjustment function
    - Modified Cycle 1 shutdown spike adjustment function
  - STME:
    - Band-model predictions on scaled plumes (0–160 kft).
    - Estimated afterburning increase
    - Estimated base burning radiation
    - Estimated plume interference effects
- ***CONVECTION***
  - PLUME INTERACTIONS: From preliminary plume studies
  - INCIPIENT RECIRCULATION: Based upon engine spacing empirical study
  - CHOKED FLOW ALTITUDE: Empirical, TND-1093
  - STME RECIRCULATION: From scaled data base (Shuttle Orbiter, Saturn V S-11 Stage S-I S-IV Stage)
  - ASRB RECIRCULATION: From Shuttle data base and ASRB Cycle 1 methodology

NLS LOW ALTITUDE BASE BURNING ANALYSIS OBJECTIVES

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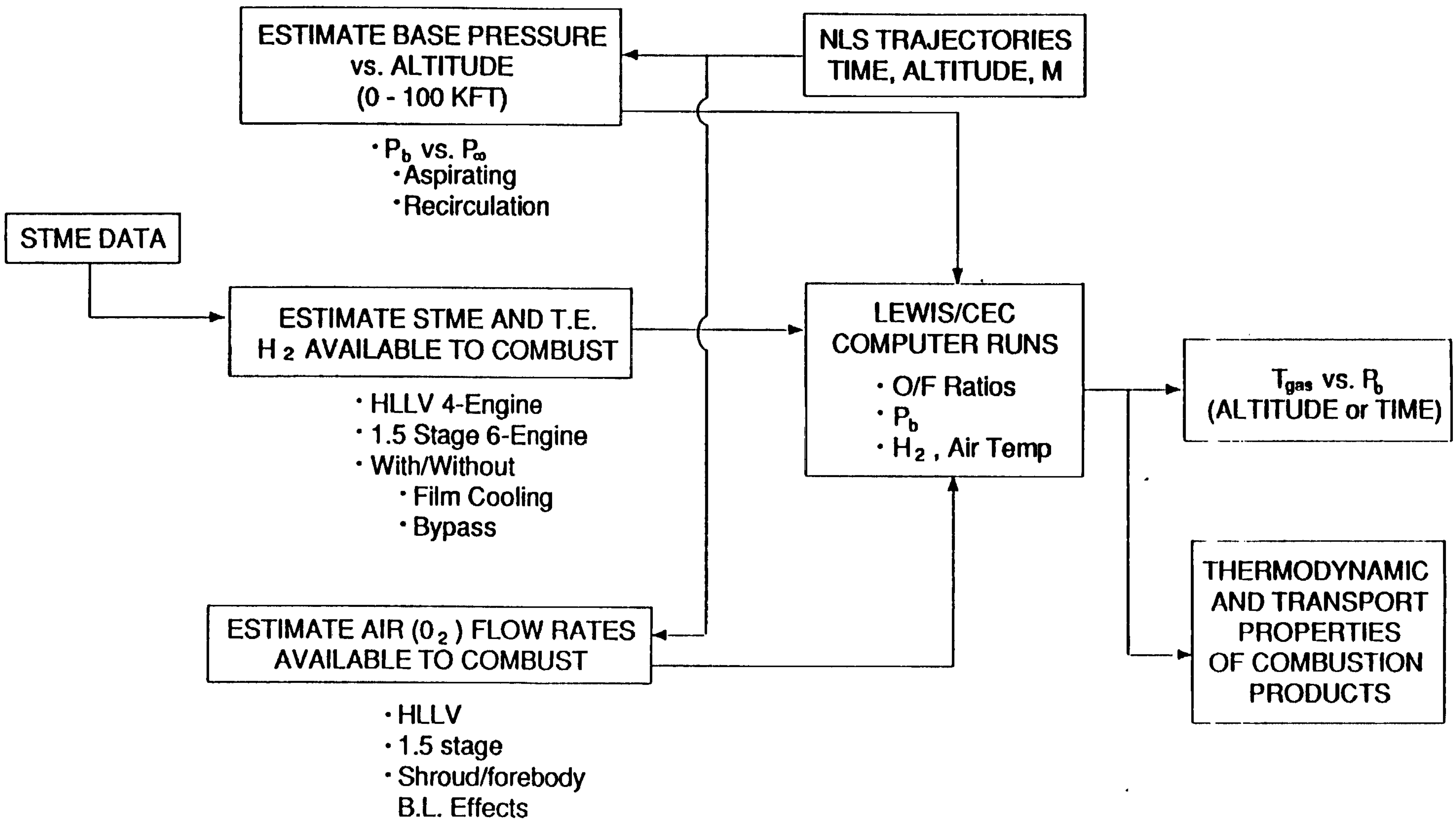
$$Q_c = hc\,(T_{gas} - T_{wall})$$

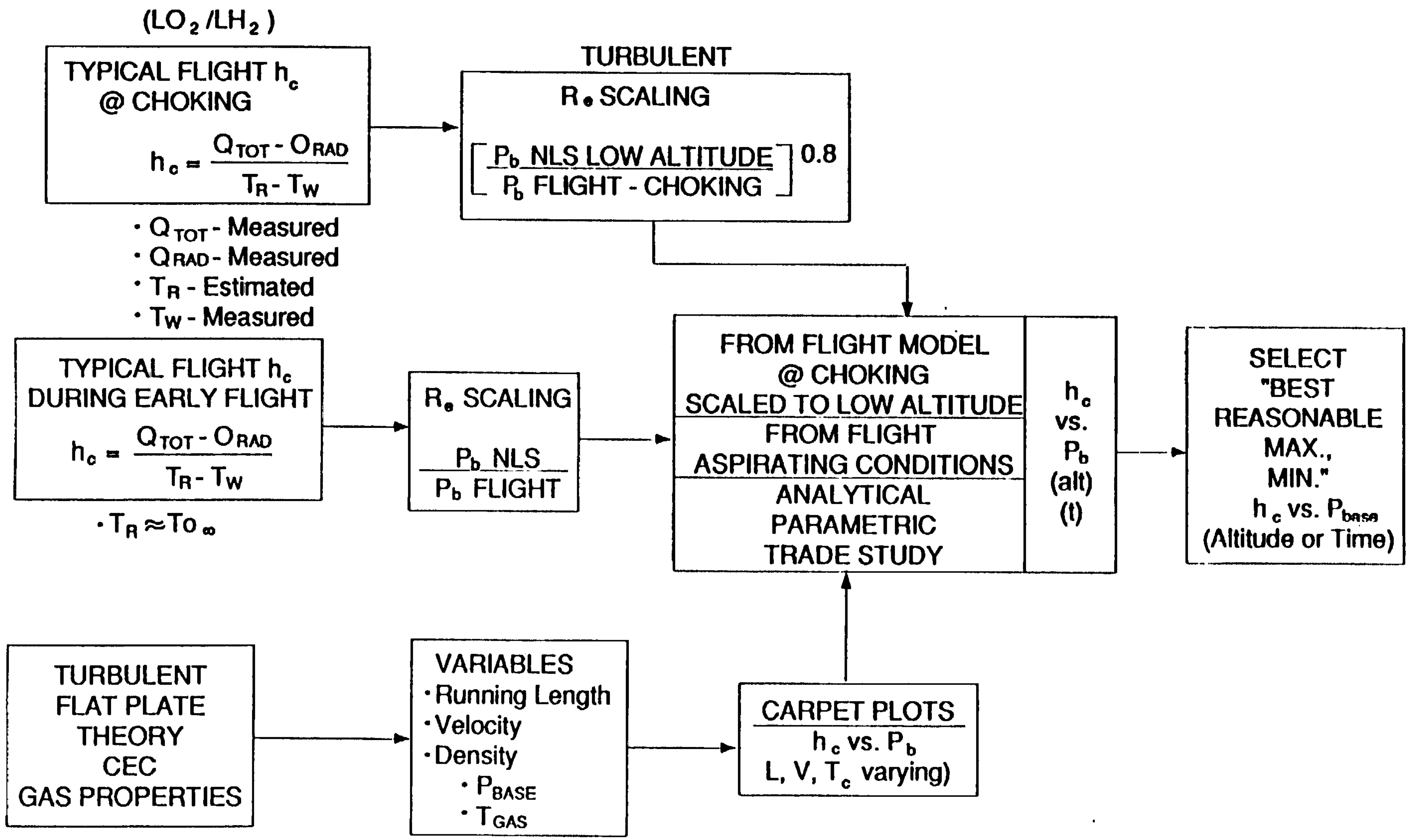
- Assuming H_2 from STME exhaust and turbine exhaust recirculatedd into base region and combusted with air at low altitudes

- The analysis will:
  - Define base region gas recovery temperature
  - Define convective heat transfer coefficient
  - Define upper altitude limit for H_2 air combustion
  - Compute convective heating rate
    - Base heat shield
    - STME heat shield
    - STME nozzle exterior

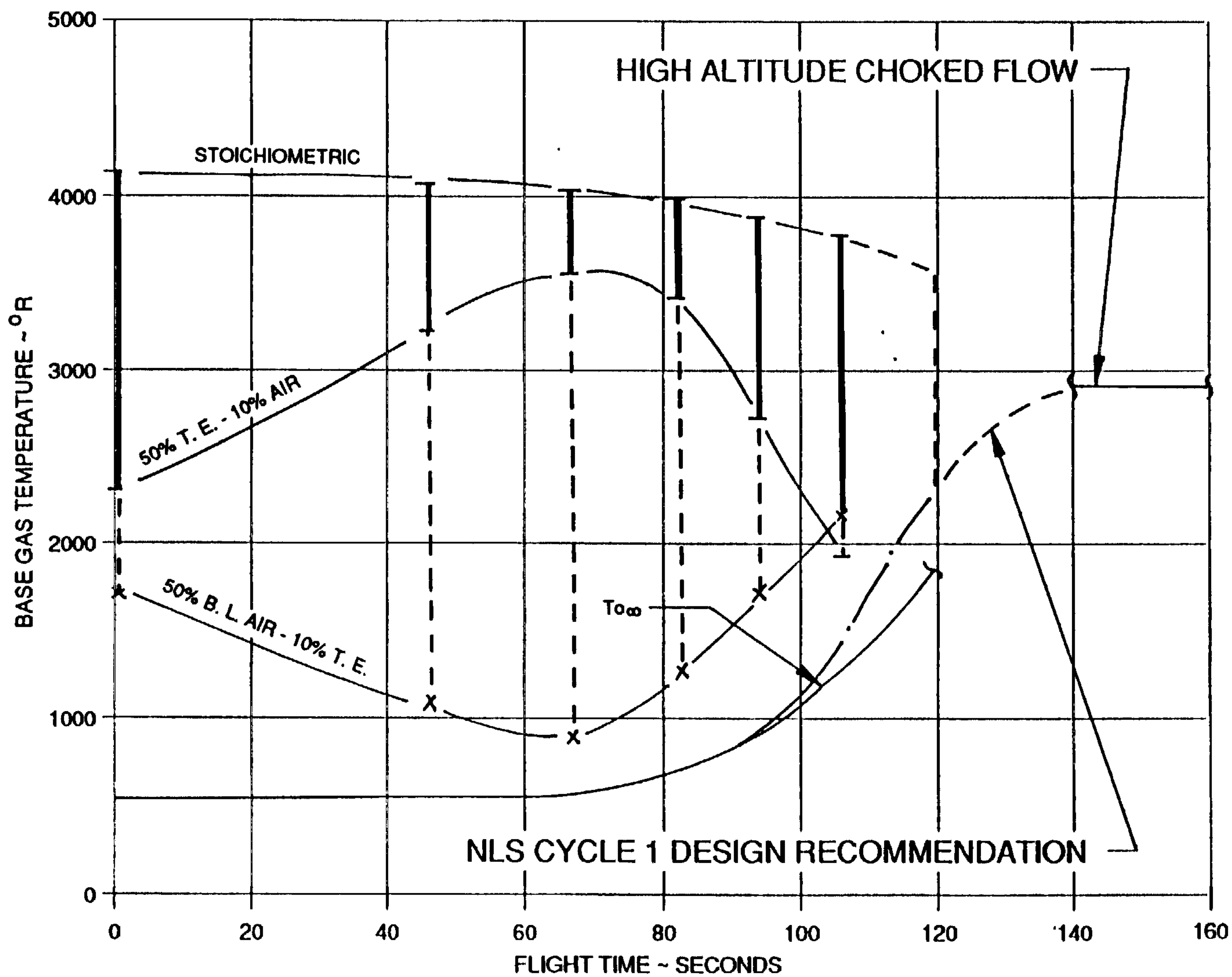
DETERMINATION OF GAS RECOVERY TEMPERATURE



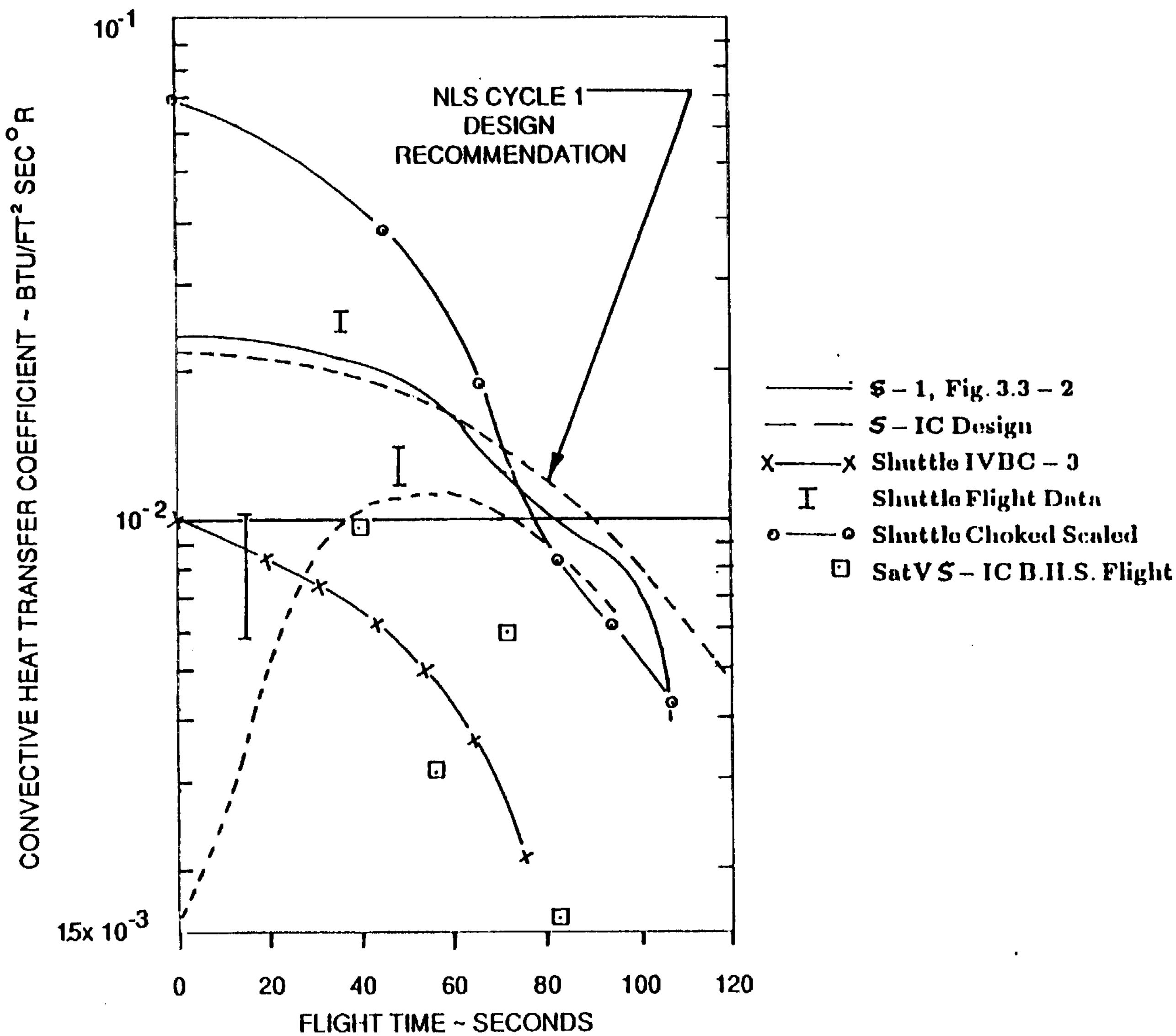
DETERMINATION OF CONVECTIVE HEAT TRANSFER COEFFICIENT



NLS - ESTIMATED CONVECTIVE BASE HEATING WITH TURBINE EXHAUST BASE BURNING



NLS - CONVECTIVE HEAT TRANSFER COEFFICIENT ESTIMATES FOR CORE BASE REGION



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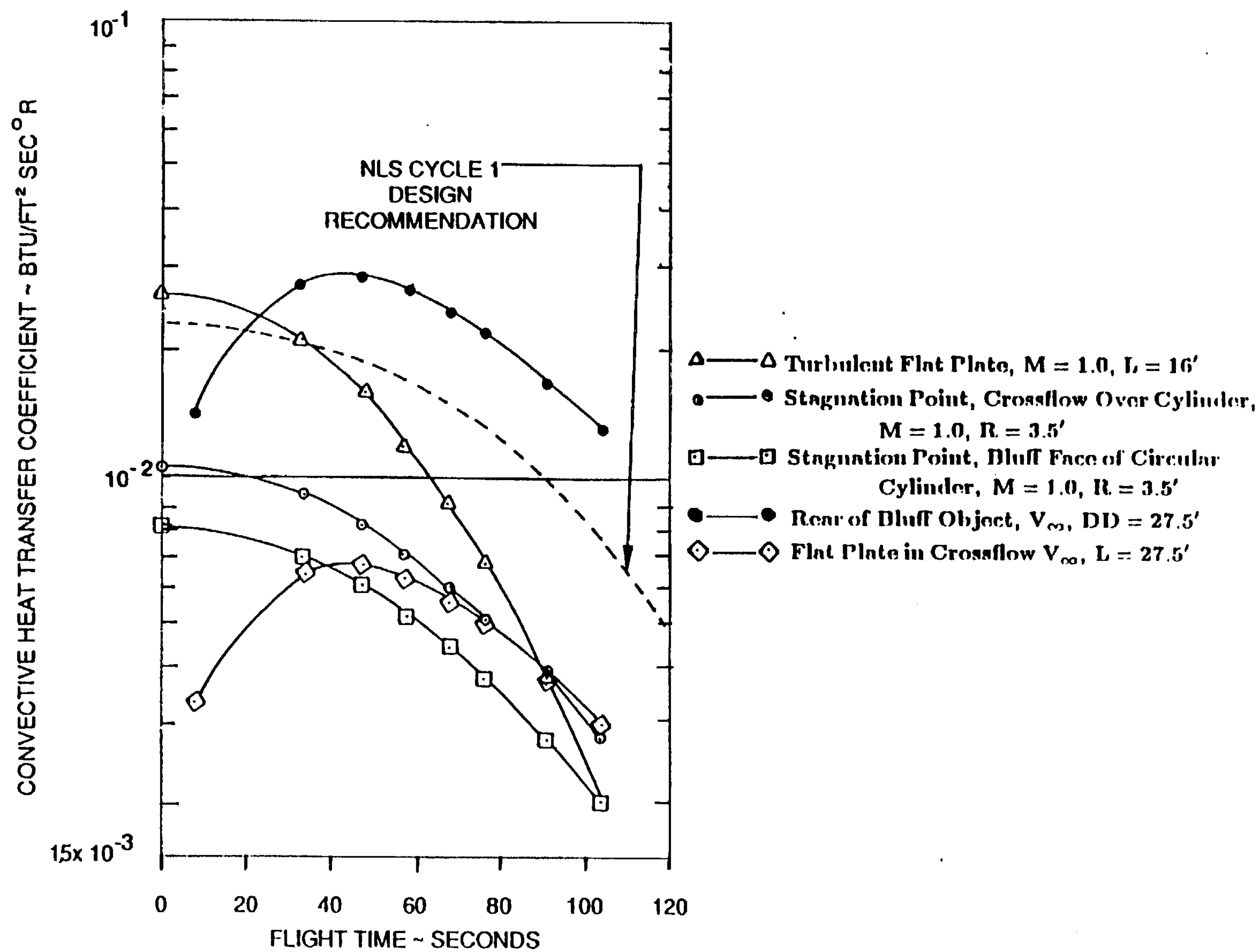

RESULTS OF BASE BURNING ANALYSIS

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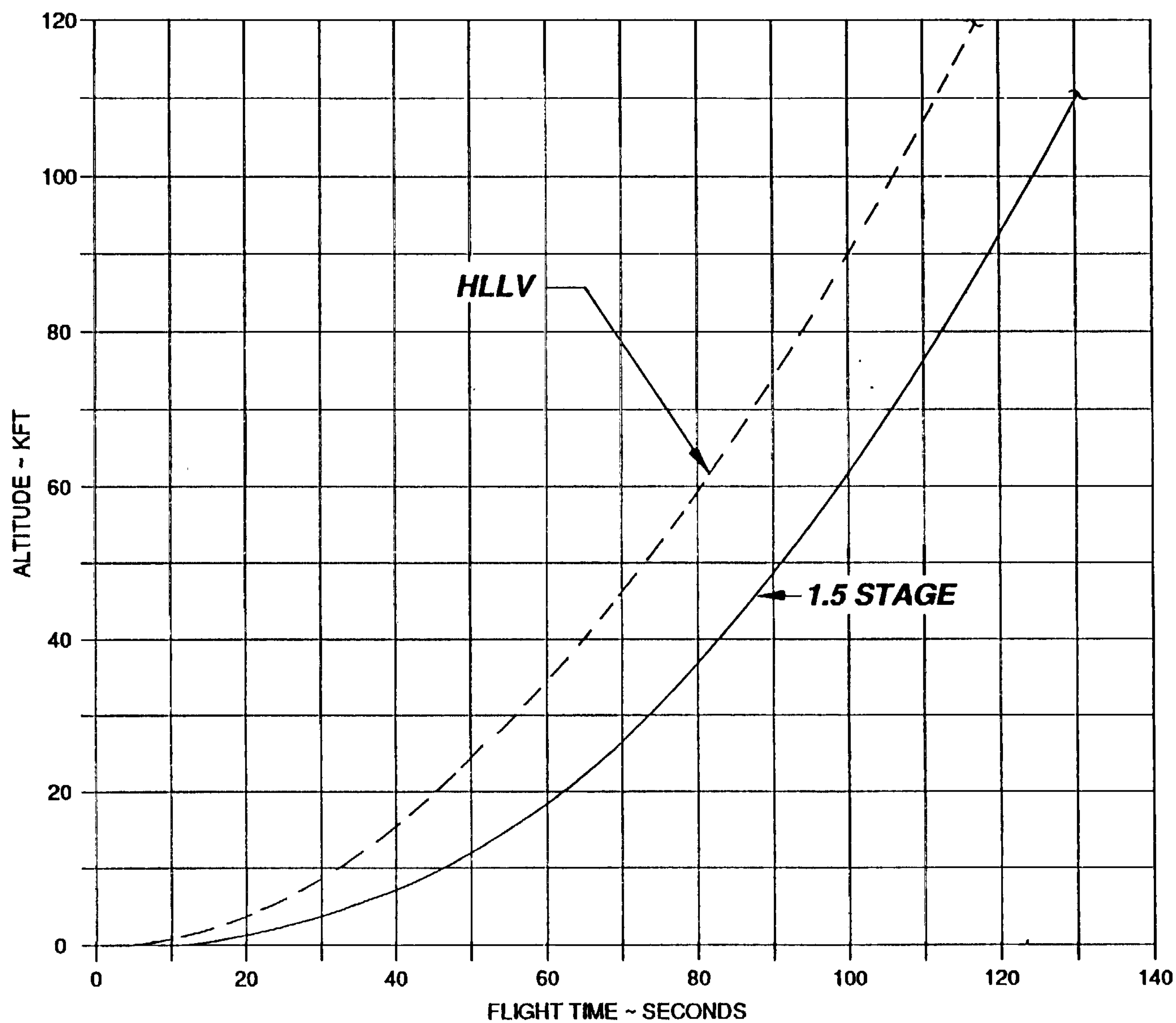
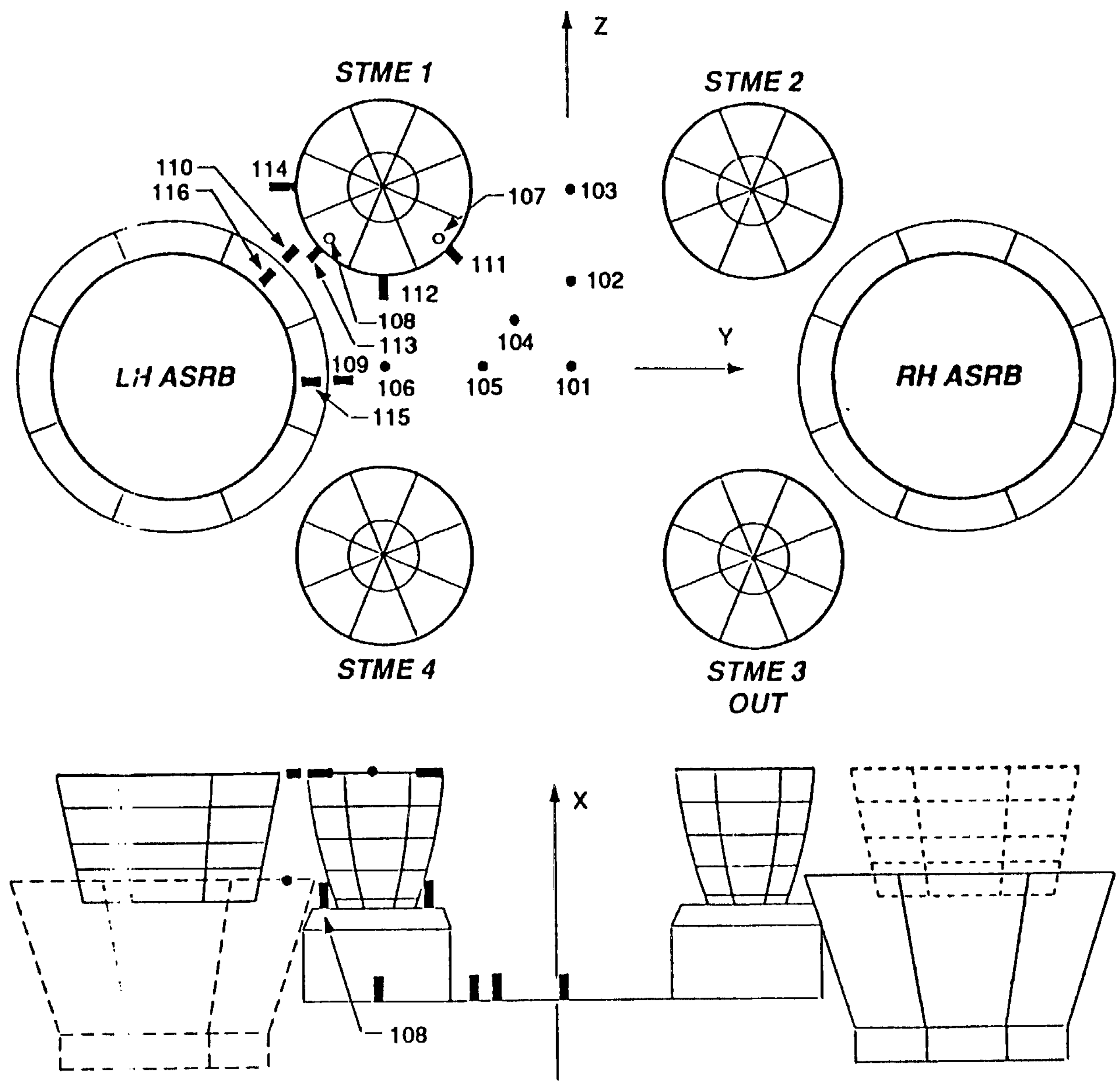
- Complex NLS base flowfields can recirculate low energy STME nozzle exhaust into base region at any altitude.
- Low energy plume boundary gases near nozzle lip will contain significant quantity of unburned H_2 and H_2O with current STME turbine exhaust disposal scheme.
- Burning of recirculate H_2 with air in base can occur from sea level to approximately 120,000 feet.
- Base gas temperatures as a result of H_2 burning can approach 4000° R at low altitudes.
- Convective heat transfer coefficients on the order of 2×10^{-2} BTU/ft^2 sec °R are feasible in the base at typical low altitude densities and turbulence levels.
- Convective heating rates as high as 80 BTU/ft^2 sec (cold wall) are possible.

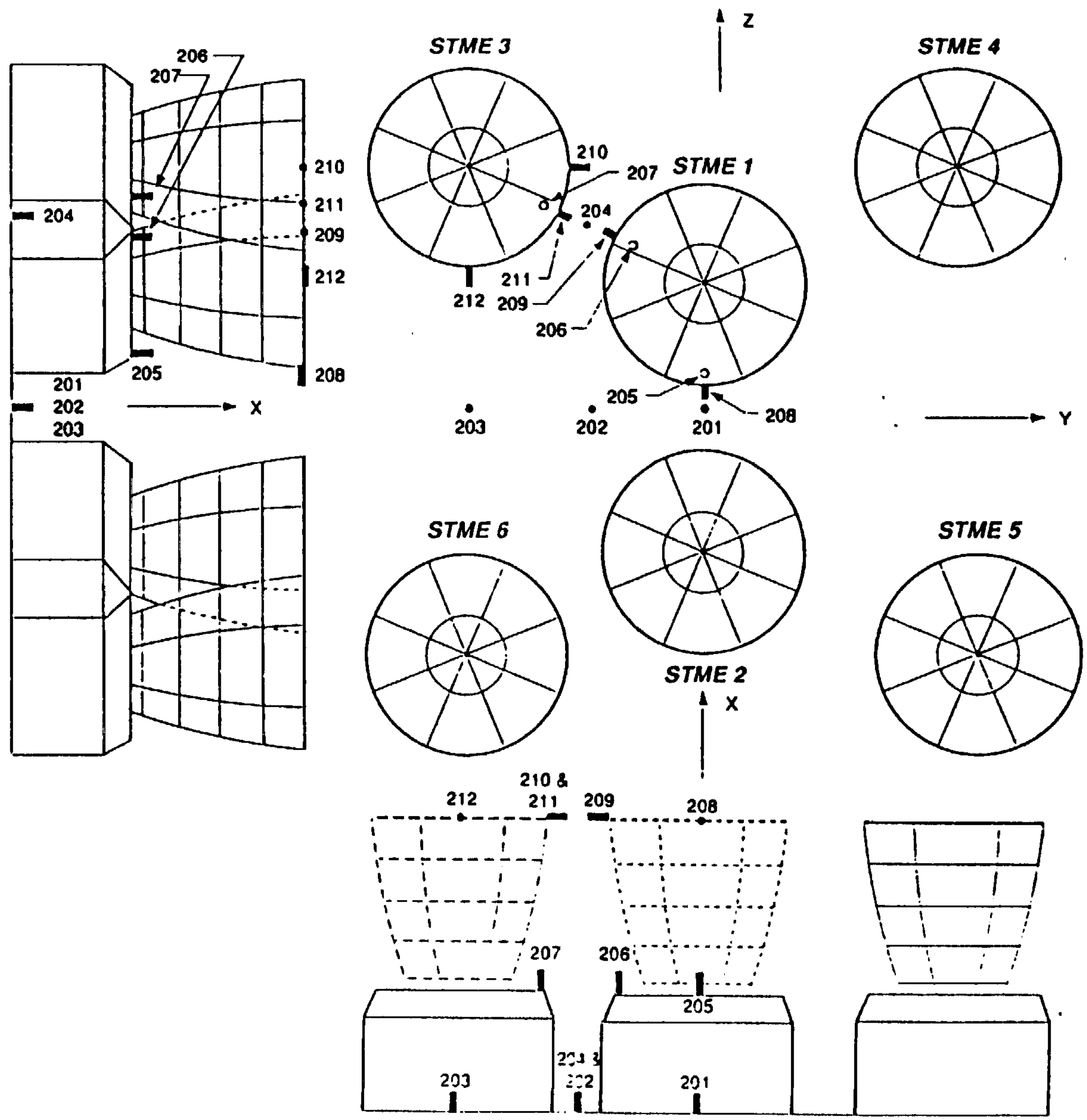
NLS BASE HEATING TRAJECTORY - ALTITUDE vs TIME



HLLV BODY POINTS SELECTED FOR BASE HEATING ANALYSIS



1.5 STAGE BODY POINTS SELECTED FOR BASE HEATING ANALYSIS



NLS RADIATION ENVIRONMENTS AT SEA LEVEL

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HLLV RADIATION ENVIRONMENTS

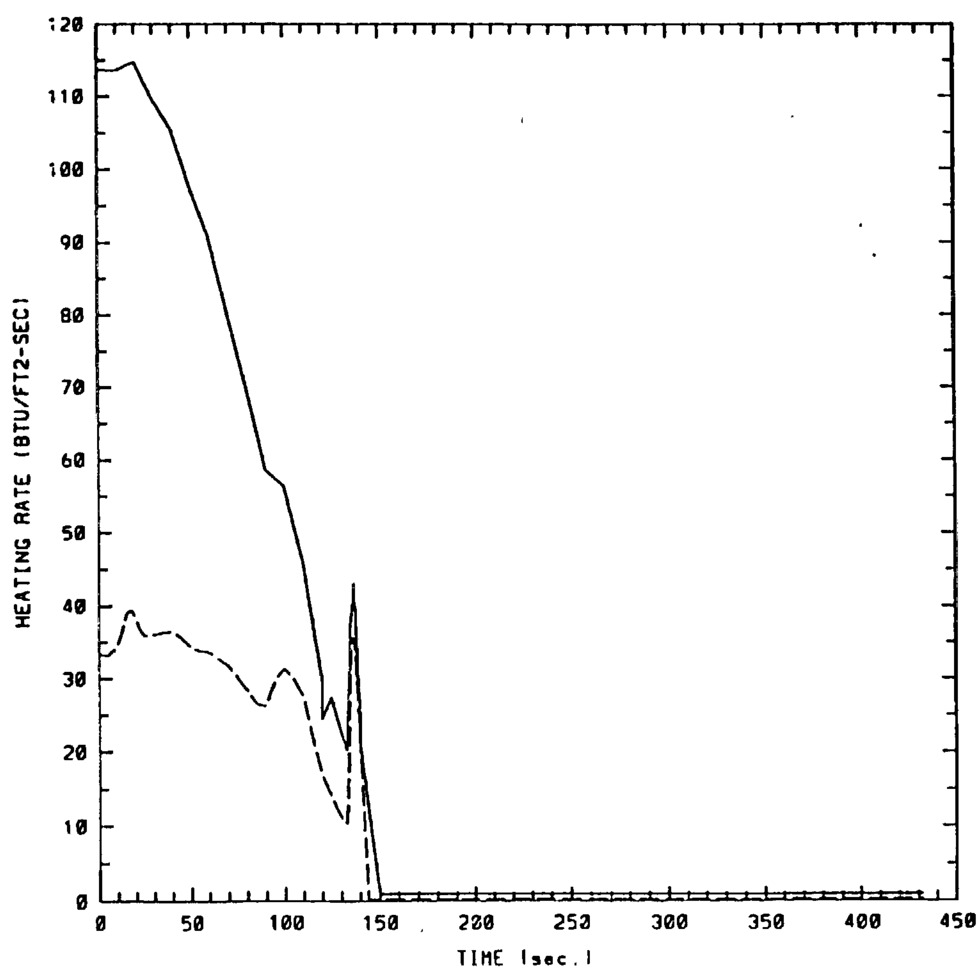
TIME (SEC)	HEATING RATE (BTU/FT SEC) FOR POINTS LISTED															
	101	102	103	104	105	106	107	108	109	110	111	112	113	114	115	116
0.0	26.52	25.56	20.51	25.48	25.65	21.24	23.69	21.08	7.15	2.27	17.27	20.83	33.39	17.59	19.28	15.26

1.5 STAGE RADIATION ENVIRONMENTS

TIME (SEC)	HEATING RATE (BTU/FT SEC) FOR POINTS LISTED											
	201	202	203	204	205	206	207	208	209	210	211	212
0.0	4.98	8.30	11.27	5.18	9.82	5.90	12.36	13.48	6.72	12.73	17.57	10.51

HLLV BASE HEATING ENVIRONMENTS AT B.P. 113

Radiation and Total Base Heating — HLLV STME Nozzle Exit Body Point 113



NLS CYCLE 1 ENVIRONMENT CONCLUSIONS

HLLV:

- Radiation dominated by ASRB plume radiation
- Predicted radiation from base gas burning is negligible
- Maximum radiation rate of 39.4 $BTU/ft^2 sec$ predicted for STME nozzle exit
- Convective heating resulting from turbine exhaust burning dominates the total environment during the first 120 seconds of ascent
- Maximum convective rate of 80.3 $BTU/ft^2 sec$ is predicted near sea level
- Core vehicle convection after ASRB separation is minimal

1.5 STAGE:

- Convective trajectory used becaue of difficulty in dealing with time mismatch; minor effect of radiation compared with convection
- Small increase caused by throttling (lower plume expansion)
- Altitude and magnitude of radiation from base burning very approximate
  Maximum radiation rate of 17.6 predicted for STME nozzle exit
- Convective heating due to turbine exhaust burning will dominate the environment during the first 120 seconds of ascent
- Maximum convective rate of 80.3 is predicted near sea level

IMPLICATIONS OF PROPOSED STME DESIGN CHANGES

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- Upgrading current STME design to 650K has small impact (approx. 5 to 10% increase) on Cycle 1 environments.

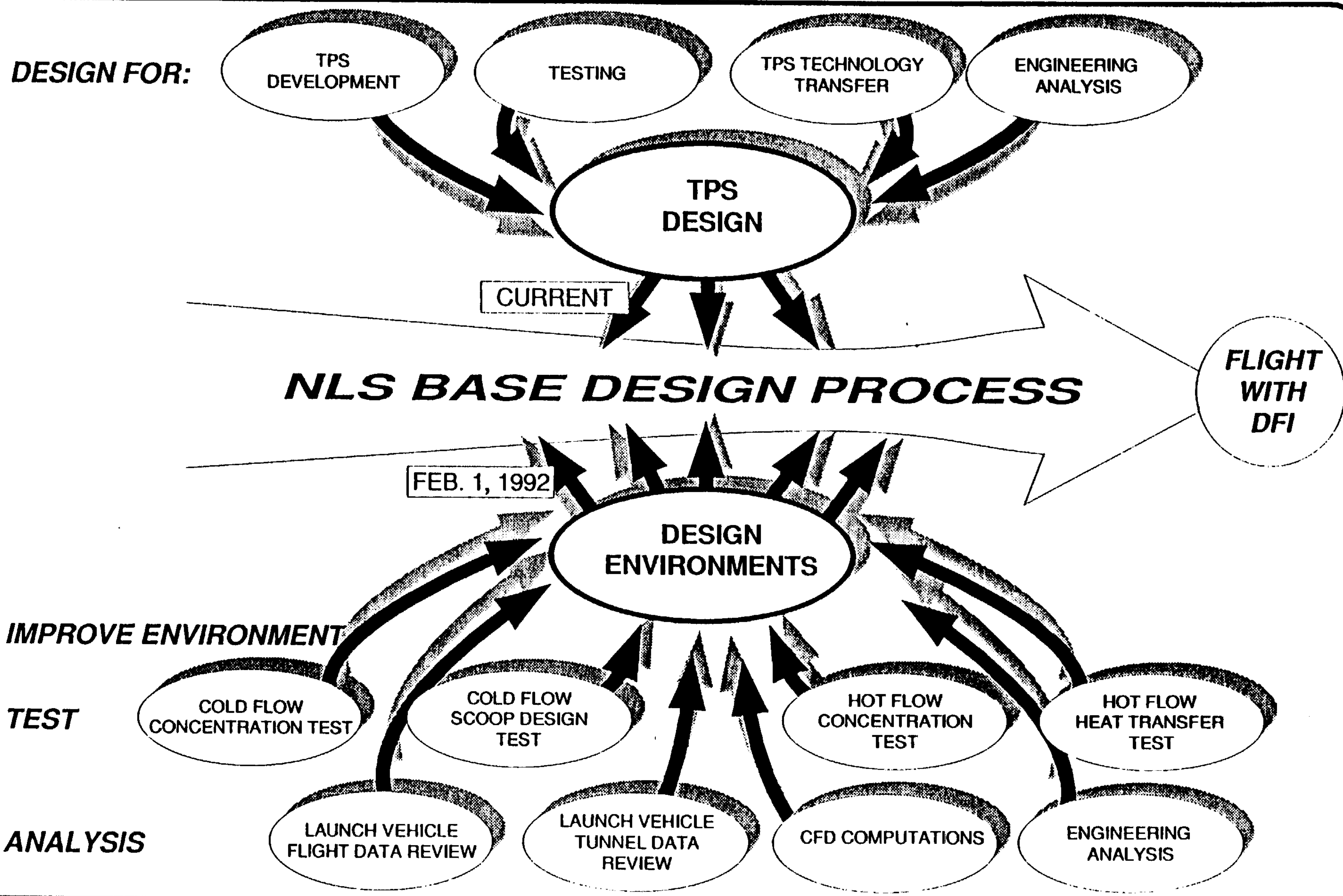
- If STME remains G.G. cycle engine:
  1) Variations in nozzle disposal schemes have little impact on current conservative base burning analysis approach and resulting environments.
  2) Outboard ducts change base burning potential - but have not been analyzed.

- Regenerative cooled dual combustion engine similar to SSME would effectively eliminate low altitude base burning

VERY NEAR TERM FOLLOW-ON ANALYSIS PLAN

- Revisit flight data
  - Saturn 1 Block I (SA-1 through SA-4)
  - Saturn V/S-1C Stage (501 through 505)
- Reconstruct heat transfer coefficient envelopes from flight data
- Adjust envelopes to NLS flight conditions
- Reevaluate choice of "original" Saturn I flight deduced heat transfer coefficient for NLS design environment
- Propose new coefficient, if indicated

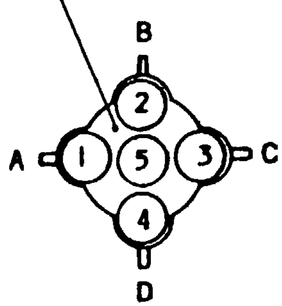
GLOBAL APPROACH

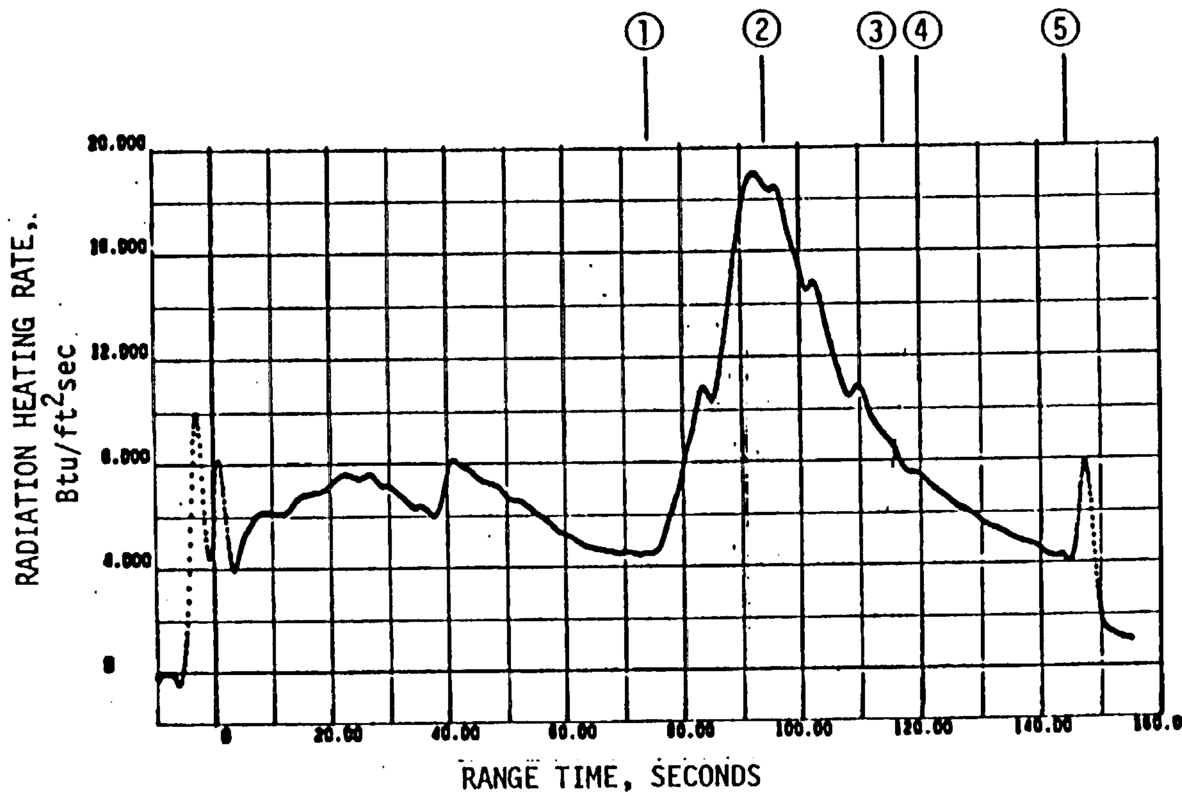


COMPARISON OF FLIGHT DATA AND TV CAMERA COVERAGE, AS-502 FLIGHT DATA

OBSERVED EVENT
TV CAMERA COVERAGE

1. FIRST FLAME (RECIRCULATION)
2. FULL RECIRCULATED FLOW IN BASE REGION
3. HEAT SHIELD BLACKENED
4. AREA BETWEEN ENGINES BECOMES CLEAR
5. BRIGHTENING AT CENTER ENGINE CUTOFF





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RADIATION SOURCES

- LOW ALTITUDE (< 70 kft)
  - * Plume Core (Mach Disk)
  - * Afterburning
  - * Baseburning (Turbine Exhaust)
- HIGH ALTITUDE (> 70 kft)
  - * Plume Core (Near Field)
  - * Plume Interaction Zones
  - * Base Recirculation
- SRM SHUTDOWN SPIKE

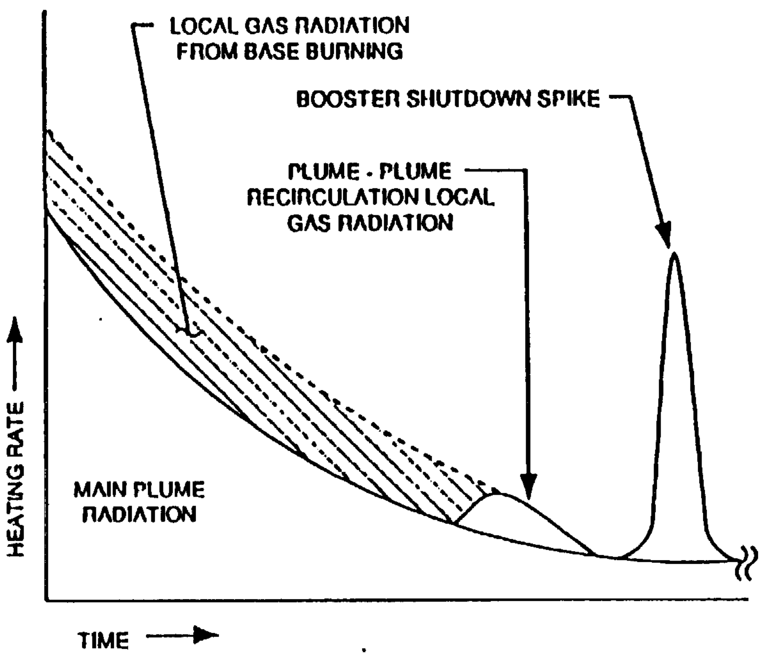
CONVECTION SOURCES

- COOLING FROM AMBIENT AIR
- HEATING FROM RECIRCULATED PLUME GASES
  - * PLUME-PLUME INTERACTIONS
  - * PLUME-FREESTREAM INTERACTIONS
- BASE BURNING FROM RECIRCULATED TURBINE EXHAUST

BASE BURNING vs CONVENTIONAL BASE HEATING

RADIATION

- Base burning increase in radiation normally small compared with conventional radiation



CONVECTION

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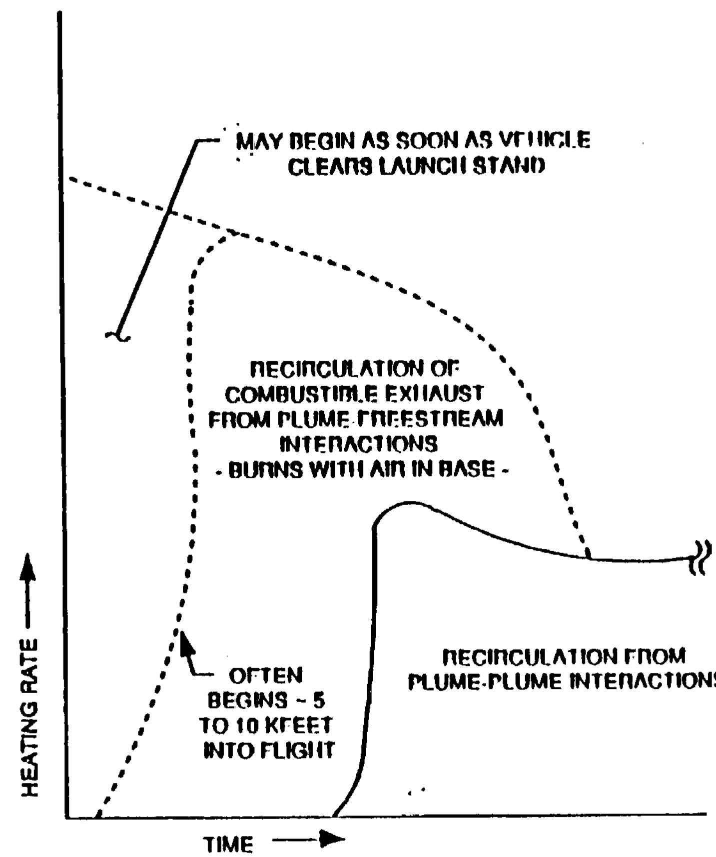


HOW DOES TURBINE EXHAUST DISPOSAL AFFECT BASE HEATING?

- If turbine exhaust dumped outboard or downstream
  - Combustible gases will burn in downstream plume and are not entrained in local recirculation pattern.
  - Amount of combustible exhaust product in engine nozzle boundary layer is small — so base region convection due to recirculated gases is determined by nozzle boundary layer gas temperature.
  - Afterburning in near plume and resultant change in plume radiation is minimized.
- If turbine exhaust dumped directly in base, engine nozzle, or nozzle exit plane.
  - Local combustion of turbine exhaust gases will occur in base region when oxidizer is present and base pressure is sufficient — referred to as base burning.
  - Base burning increases base gas temperature, alters base flow patterns, and may dramatically increase base region convection and local gas radiation.
  - Nozzle injection and subsequent afterburning changes plume radiation characteristics, often increasing downstream plume radiation.

PAST EXPERIENCE WITH TURBINE EXHAUST DISPOSAL
--- LARGE U.S. LAUNCH VEHICLES ---

VEHICLE	T.E. DISPOSAL SCHEME	EXPERIENCE/LESSON LEARNED
JUPITER -1A	• Duct Along Nozzle to Exit Plane • Change to Outboard Duct	• 1st Flight Failed Due to Base Heating • No failure
ATLAS	• Duct into Base - By Center Engine • Change to Outboard Duct	• 1st 2 Flights Failed Due to Base Heating • No Failure
DELTA	• Duct through Heat Shield	• High local heating on heat shield while SRM's attached
TITAN II	• Two ducts exiting slightly aft of boattail base. • Strong air scooping eliminates base burning.	• Heating not severe • No failure due to T.E. burning
TITAN III (Core)	• Core engine ignited at H $\geq$ 100 kft; above altitude of serious burning.	• No trouble
SATURN I	• Inbd engine ducted to fin outbd of base • Outbd engine into nozzle through exhausterator.	• High heating early in flight • No failure due to T.E. burning
SATURN IB	• Inbd engine ducted through 4 crescent opening in flame shield • Exhausterator on outbd engine	• T.E. exhaust did not burn; cooled flame shield • No failure
SATURN V	S-IC Stage — F-1 Engine T.E. Dumped in Nozzle @ A/A*=10	• No Failure Due to Base Heating • Unburned RP-1 Afterburning in Plume @ Low Altitude, Burned in Base @ High Altitude
NSTS SPACE SHUTTLE	• No T.E. Disposal on SSME • SRB T.E. Dumped Outboard	• No Failure Due to Base Heating • Predictable Environments

SUMMARY OF TURBINE EXHAUST DISPOSAL
FLIGHT EXPERIENCE

- Flight vehicles with turbine exhaust disposal into base, engine nozzle, or external flow.
  - ATLAS
  - SATURN 1 & 1B, 1st Stage
  - SATURN V, 1st Stage
  - DELTA

  } LO_2/RP-1 Propellants

  - TITAN — Aerozine 50/UDMH Propellants (Storable)
- Flight vehicles which utilized LO_2/LH_2 propellants.
  - S–IV Stage, SATURN 1
  - S–II Stage, SATURN V
  - S-IV B Stage, SATURN V

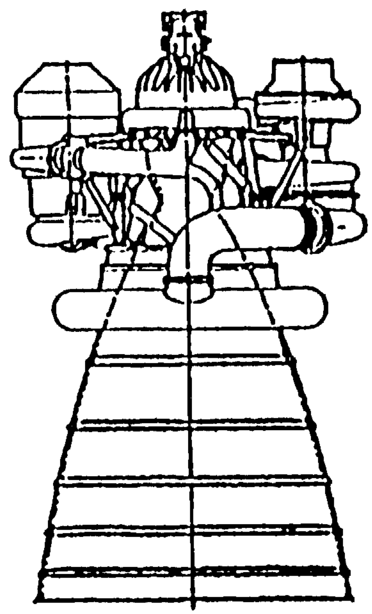
  } T.E. Dumped inside nozzle-high altitude.

  - Shuttle Orbiter — Regeneratively cooled nozzle — no T.E. Discharge

THE NLS - STME TURBINE EXHAUST DILEMMA

- The STME with film/convective dump cooled nozzle:
  - is a new concept, outside experience range
  - creates potential for large mass flow of low energy, unburned H_2 at nozzle exit lip
  - H_2 will burn over wide range of mixture ratios (and pressures) with oxygen (air) present in base.
- Both NLS configurations have complex base flowfields and potential for low altitude recirculation
  - HLLV — close proximity of ASRB (with skirt) and STME (with shroud).
  - 1.5 Stage — close proximity of sustainer engines and sustainer/booster engines

STME FILM/CONVECTIVE DUMP COOLED NOZZLE



Thrust, lbs	583,000
Chamber Pressure, psia	2250
Mixture Ratio	6.0
Min. Specific Impulse (vac) sec	430.5
Weight, lbs	8000
Area Ratio	45

MAIN CHAMBER

$P_o - 2250^{\circ}\ psia$

$T_o - 6708^{\circ}\ R$

$\dot{\omega} - 1292.7\ lbm/sec$

TURBINE EXHAUST DISCHARGE

•Primary Film Coolant

$P_o - 204\ psia$

$T_o - 1190^{\circ}\ R$

$\dot{\omega} - 24.4\ lbm/sec$

•Secondary Film Coolant

$P_o - 80.3\ psia$

$T_o - 1190\ R$

$\dot{\omega} - 4.26\ lbm/sec$

•Convective Coolant

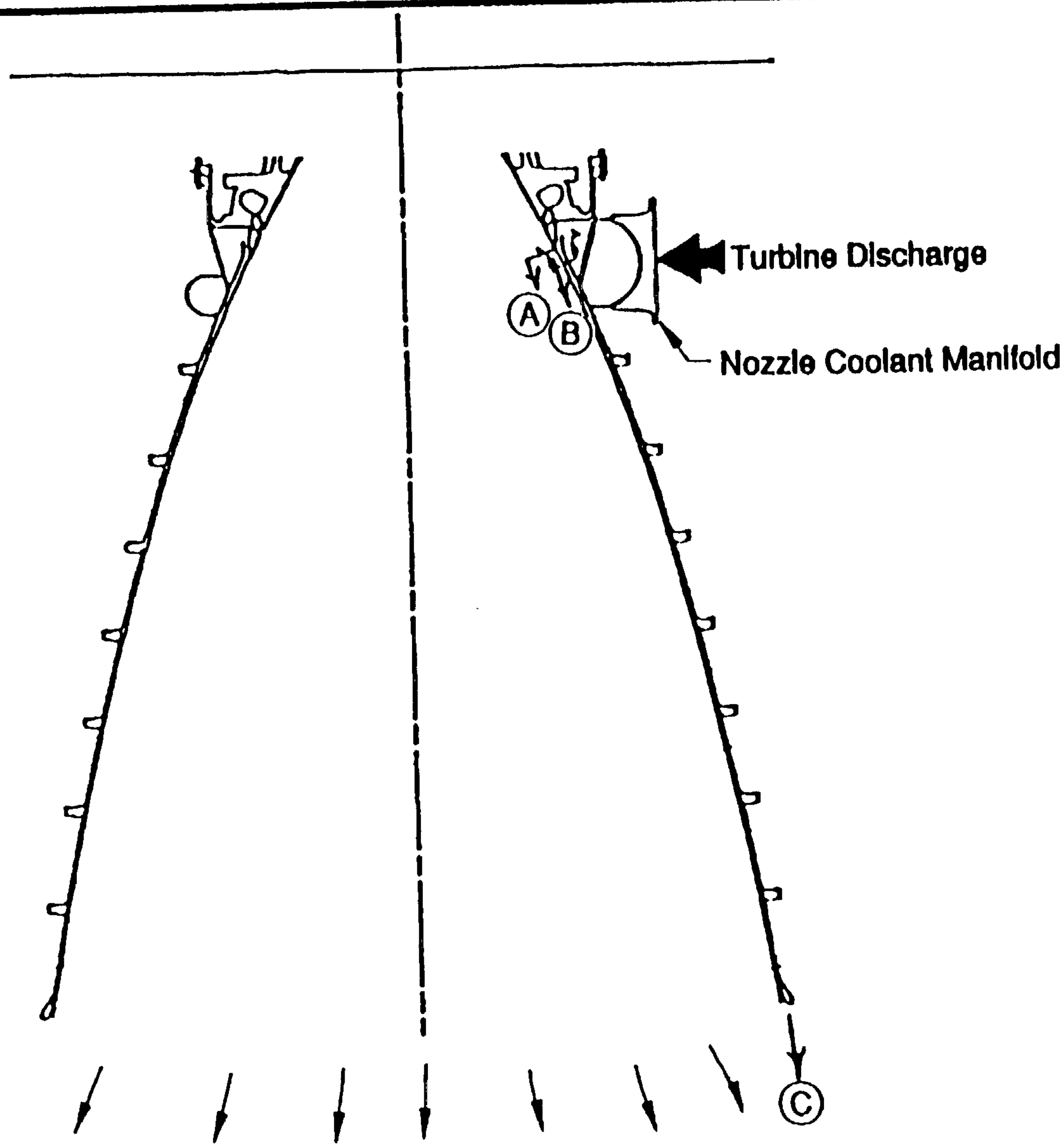
$P_o - 88.8\ psia$

$T_o - 1462.4^{\circ}\ R$

$\dot{\omega} - 35.4\ lbm/sec$

NOTE: Turbine exhaust is: 47% H_2
53% H_2O (Steam)

STME HYDROGEN FLOW RATES



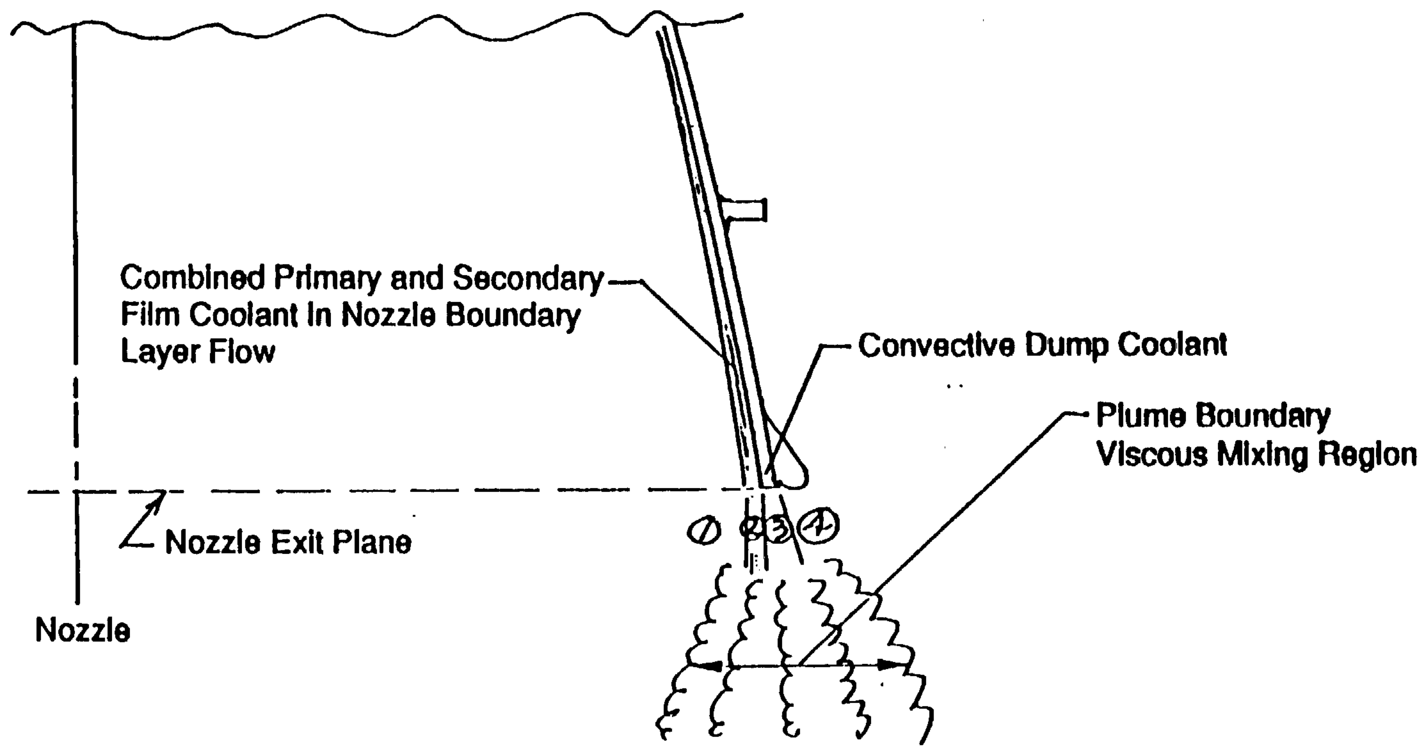
MAIN ENGINE FLOW: $16.7\ lbm\ H_2/sec\ @\ O/F = 7.1$

(A) ***Secondary Film Coolant***
2.0 lbm H_2/sec.

(B) ***Primary Film Coolant***
11.4 lbm H_2/sec.

(C) ***Convective Dump Coolant***
16.5 lbm H_2/sec.

STME PLUME EXPANSION/RECIRCULATION FLOWFIELD



<u>Four (4) Stream Mixing Problem</u>

1) Nozzle Inviscid Flow - $P_o \approx 2200$ psia
2) Film Coolant/Nozzle Boundary Layer Flow - $P_o \approx 200$ psia
3) Convective Dump Coolant Flow - $P_o \approx 90$ psia
4) Freestream or Base Region Flow - $P_o \approx 14.7$ or less psia

F-1 ENGINE/STME COMPARISONS

Comparison Parameter	F-1 Engine	STME
Operating Conditions		
• Chamber Pressure, PSI	1126/983	2250
• Chamber Temperature (°R)	6383	6708
• Area ratio	16	45
• Propellants	LOX/RP-1	LO_2/LH_2
• O/F	2.27	7.1
• 1D Exit Pressure, PSIA	6.18	
Nozzle Description		
• Exit Diameter	140″	87.8″
• Nozzle Half Angle	13°	
Flow Rates		
Main Chamber, lbm/sec	5564.4	1292.7
Turbine Exhaust Total lbm/sec	170.5	64.06
O/F	0.42	
Turbine Exhaust, Fuel Only, lbm/sec	120.3 RP-1	29.9 H_2

F-1 ENGINE/STME COMPARISONS

Comparison Parameter	F-1 Engine	STME
Ratios		
1. $\frac{\text{Total Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0306	0.0496
2. $\frac{\text{Combustible Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0216	0.0231
T. E. Characteristics		
• Total Pressure, PSIA	57	204/89
• Temperature, °	1465°F	1190/1462°R

SATURN V/S-1C STAGE/NLS 1.5 STAGE

COMPARISON PARAMETER	S-1C STAGE F-1 ENGINE	1.5 STAGE STME	COMMENT
Base Geometry			
1. Base Diameter, ~ Inches	396"	330.96"	
2. Length from Stage Center to Outboard Engine, ~ Inches	182"	165.5	
3. Length from Base Heat Shield to Nozzle Exit Plume, ~ Inches	227.4"	141.4"	
4. Nozzle Exit Diameter, ~ Inches	140"	87.8"	
5. Overall Vehicle Length, ~ Inches	4416	3221.3	
6. Shroud Length Below Base Heat Shield, Inches (Overhang)	63.5"	0.0	
7. Shroud Angle, ~ Degrees	15	20	
8. Outboard Shroud Height, Inches	77"	63"	
RATIOS			
9. $\frac{\text{Total Eng. Exit Area}}{\text{Base Area}}$	0.6249	0.4223	1.5 Stage Base More Open
10. $\frac{\text{Engine Length}}{\text{Base Diameter}}$	0.5742	0.4263	S-1C Engines Extend Further Aft
11. $\frac{\text{Nozzle Exit Diameter}}{\text{Engine Length}}$	0.62	0.6209	Same
12. $\frac{\text{Center to Outboard ℄ Distance}}{\text{Nozzle Exit Diameter}}$	1.3	$\boxed{1.885}$ $\frac{10.86'}{7.32'} = 1.48$ $\frac{9.92'}{7.32'} = 1.35$	Center to Outboard Larger on 1.5 Stage Center to Center ≈ same
13. $\frac{\text{Forebody Length}}{\text{Base Diameter}}$	11.15	9.73	S-1C (Saturn V) More Slender

WHY IS NLS/STME BASE BURNING PROBLEM UNIQUE?

- Although general flow patterns similar to Saturn V S-1C Stage, shroud and booster geometry, number of STMEs and STME length create ***unique*** base flow field for NLS
- Current STME disposal scheme creates 4 stream mixing problem at nozzle lip which is ***unique*** and different from H-1 and F-1 engines exhausterator and manifold/slot injection schemes
- H_2 injection pressures on STME higher than H-1 or F-1 which may enhance diffusion into main plume flow but also changes momentum and turbulence in shear mixing layer - creating unique recirculation potential
- H_2 potential for burning and high energy release from combustion uniquely different from RP-1 (Kerosene)
  - H_2 has wider combustion limit than RP-1
  - H_2 has 3 to 5 times energy release of RP-1 per lb.

  NOTE: RP-1 loses energy in soot formation
- Stoichiometric burning temperatures of H_2 slightly higher than RP-1 when burned with air at comparable pressure
- Transport properties of H_2 /air combustion products different from RP-1/air products; results in different convective heating over comparable surfaces

NLS BASE HEATING ANALYSIS

CYCLE 1 OBJECTIVE: **Define Ascent Base Heating Environments which Include**

- Latest HLLV and 1.5 Stage geometry
- Latest trajectories which maximize base heating
- Nominal plume radiation and high altitude plume recirculation convection

- PLUS -

- Radiation and convection augmentation due to base burning of STME turbine exhaust

Environments Published to Date (1/9/92)

- Preliminary Cycle 1 without base burning
  MSFC memo ED33 (98-91), Sept. 25, 1991
- Preliminary Cycle 1 with base burning
  MSFC memo ED33 (03-92), Jan. 8, 1992

Environments To Be Published

- Cycle 1 Including Updated Base Burning Analysis Results
  MSFC memo ED33 (15-92), Feb. 7, 1992

NLS CYCLE 1 BASE HEATING METHODOLOGY

PRELIMINARY CYCLE 1 METHODOLOGY

$$Q_{Total} = Q_{Rad} + Q_{Conv}$$

- ***RADIATION***
  - ASRM:
    - Viewfactor predictions using Cycle 1 sea-level plume model
    - Modified Cycle 1 altitude adjustment function
    - Modified Cycle 1 shutdown spike adjustment function
  - STME:
    - Band-model predictions on scaled plumes (0–160 kft).
    - Estimated afterburning increase
    - Estimated base burning radiation
    - Estimated plume interference effects
- ***CONVECTION***
  - PLUME INTERACTIONS: From preliminary plume studies
  - INCIPIENT RECIRCULATION: Based upon engine spacing empirical study
  - CHOKED FLOW ALTITUDE: Empirical, TND-1093
  - STME RECIRCULATION: From scaled data base (Shuttle Orbiter, Saturn V S-11 Stage S-I S-IV Stage)
  - ASRB RECIRCULATION: From Shuttle data base and ASRB Cycle 1 methodology

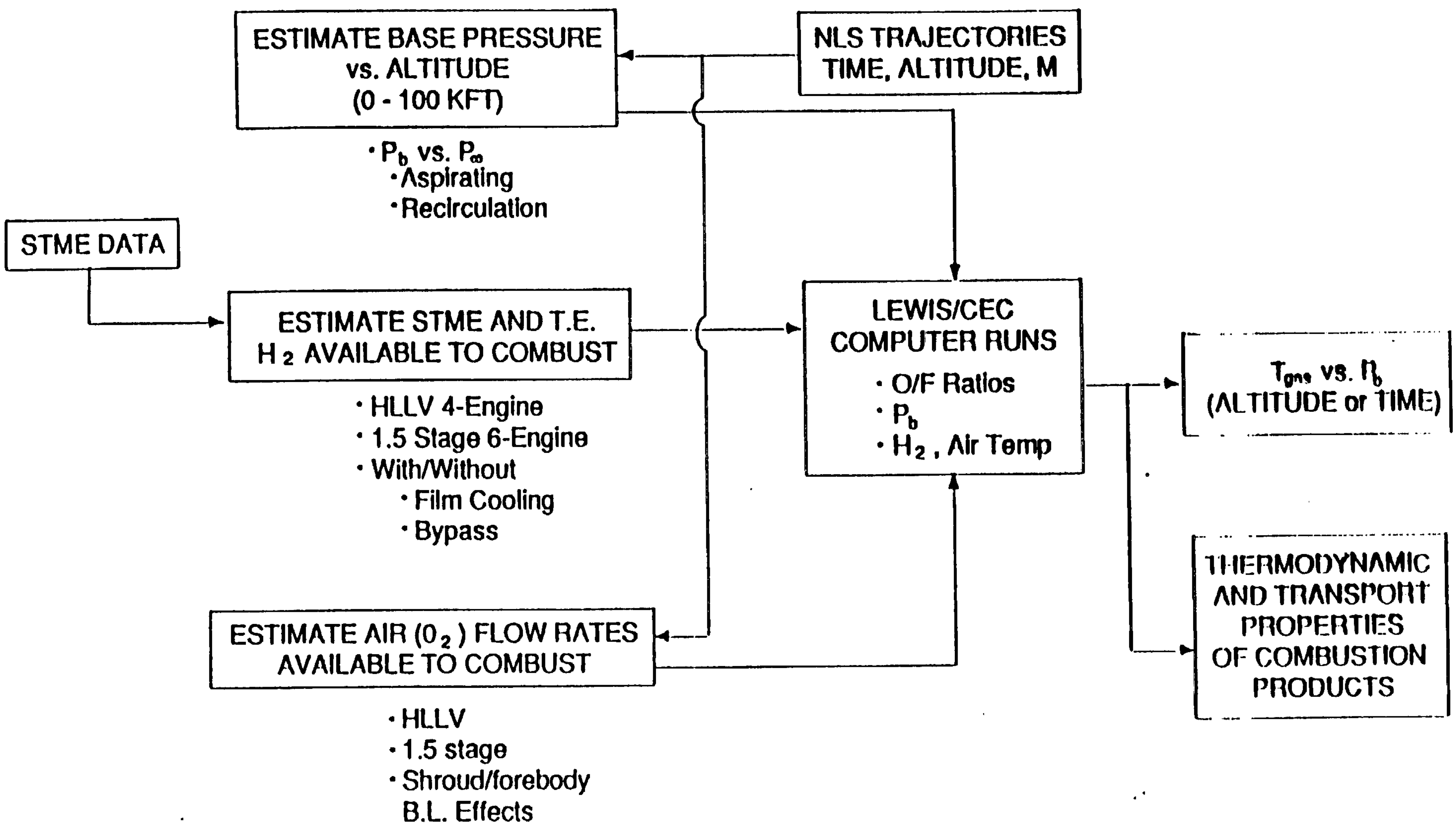
NLS LOW ALTITUDE BASE BURNING ANALYSIS OBJECTIVES

$$Q_c = hc\,(T_{gas} - T_{wall})$$

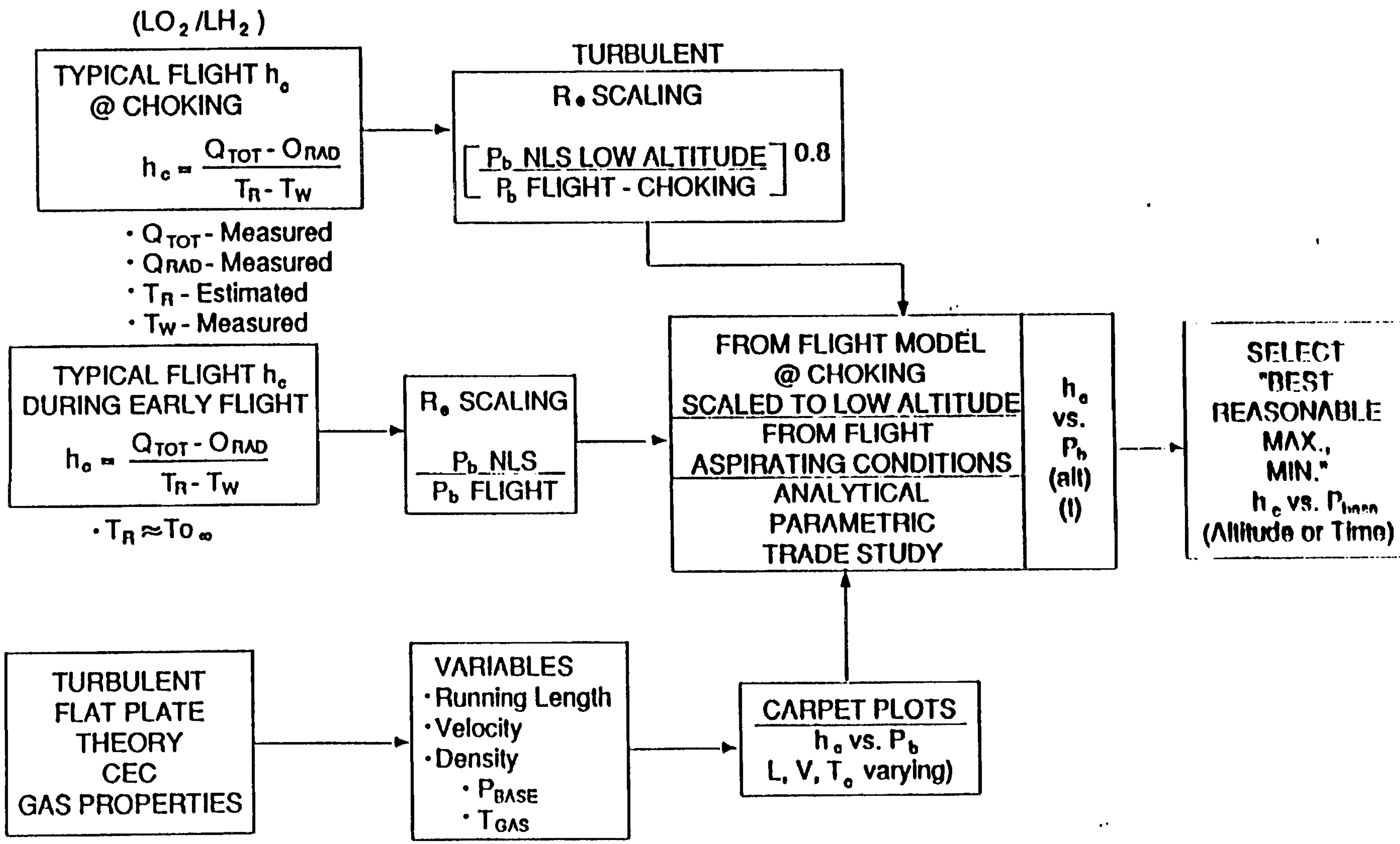
- Assuming H_2 from STME exhaust and turbine exhaust recirculatedd into base region and combusted with air at low altitudes

- The analysis will:
  - Define base region gas recovery temperature
  - Define convective heat transfer coefficient
  - Define upper altitude limit for H_2 air combustion
  - Compute convective heating rate
    - Base heat shield
    - STME heat shield
    - STME nozzle exterior

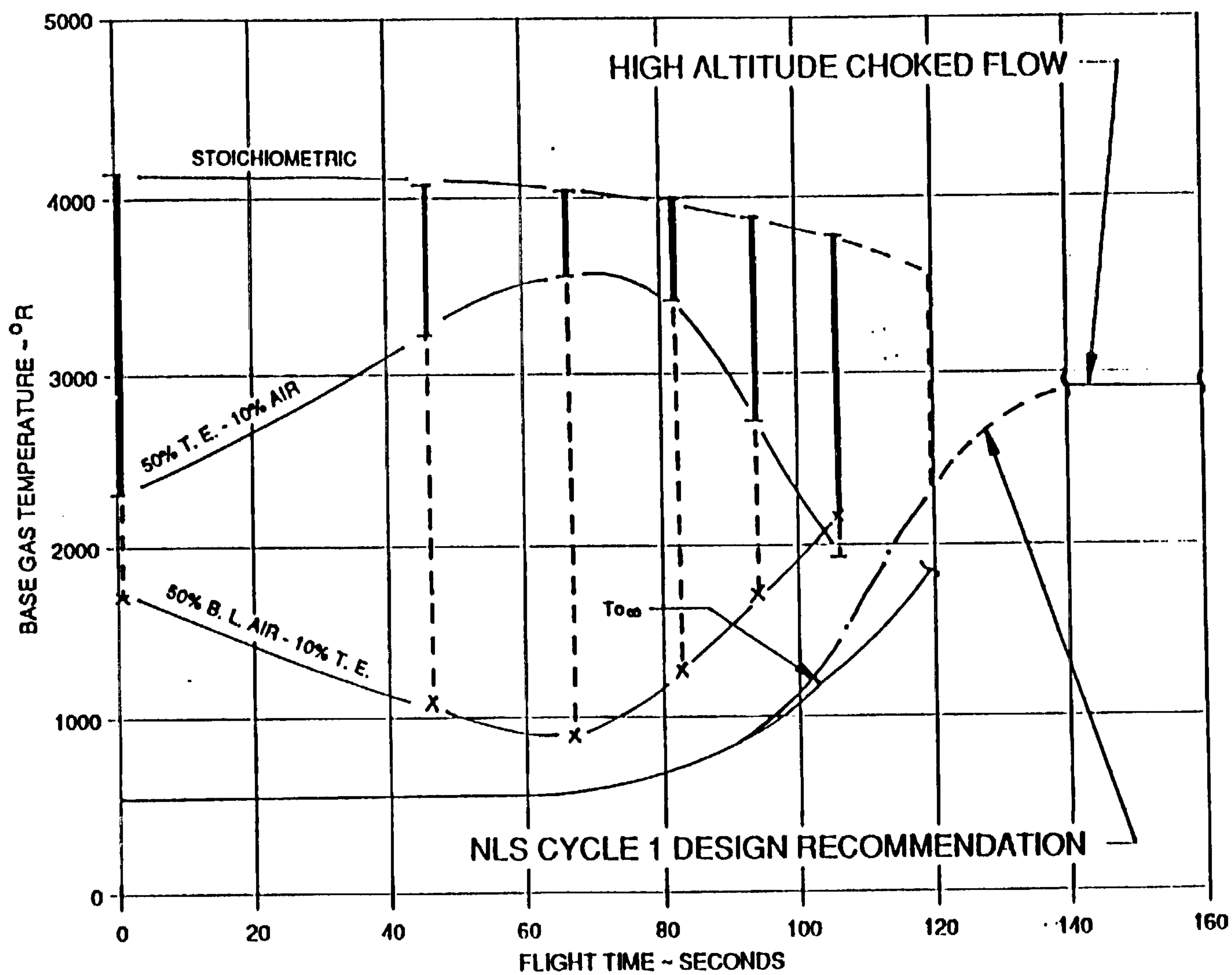
DETERMINATION OF GAS RECOVERY TEMPERATURE



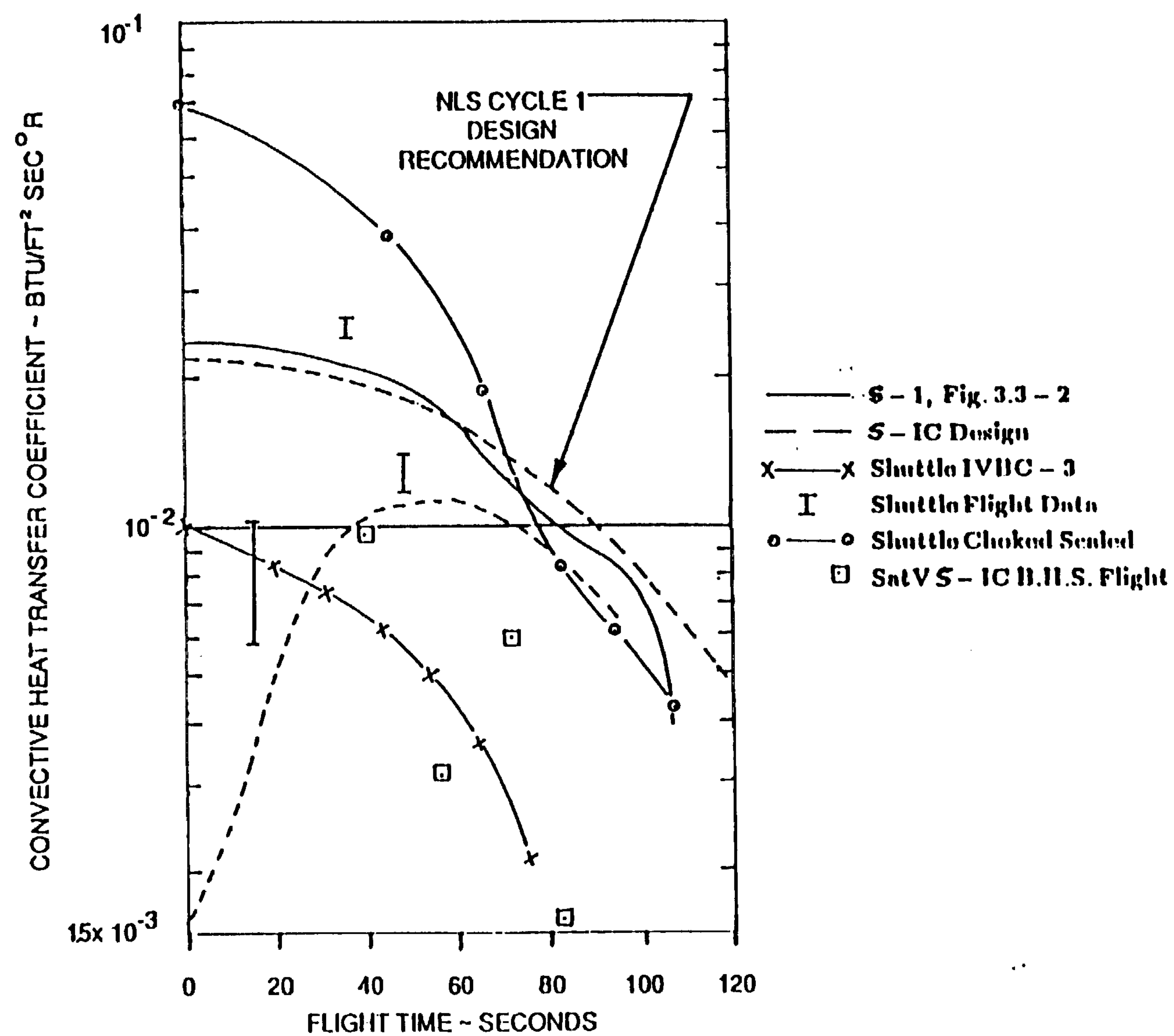
DETERMINATION OF CONVECTIVE HEAT TRANSFER COEFFICIENT



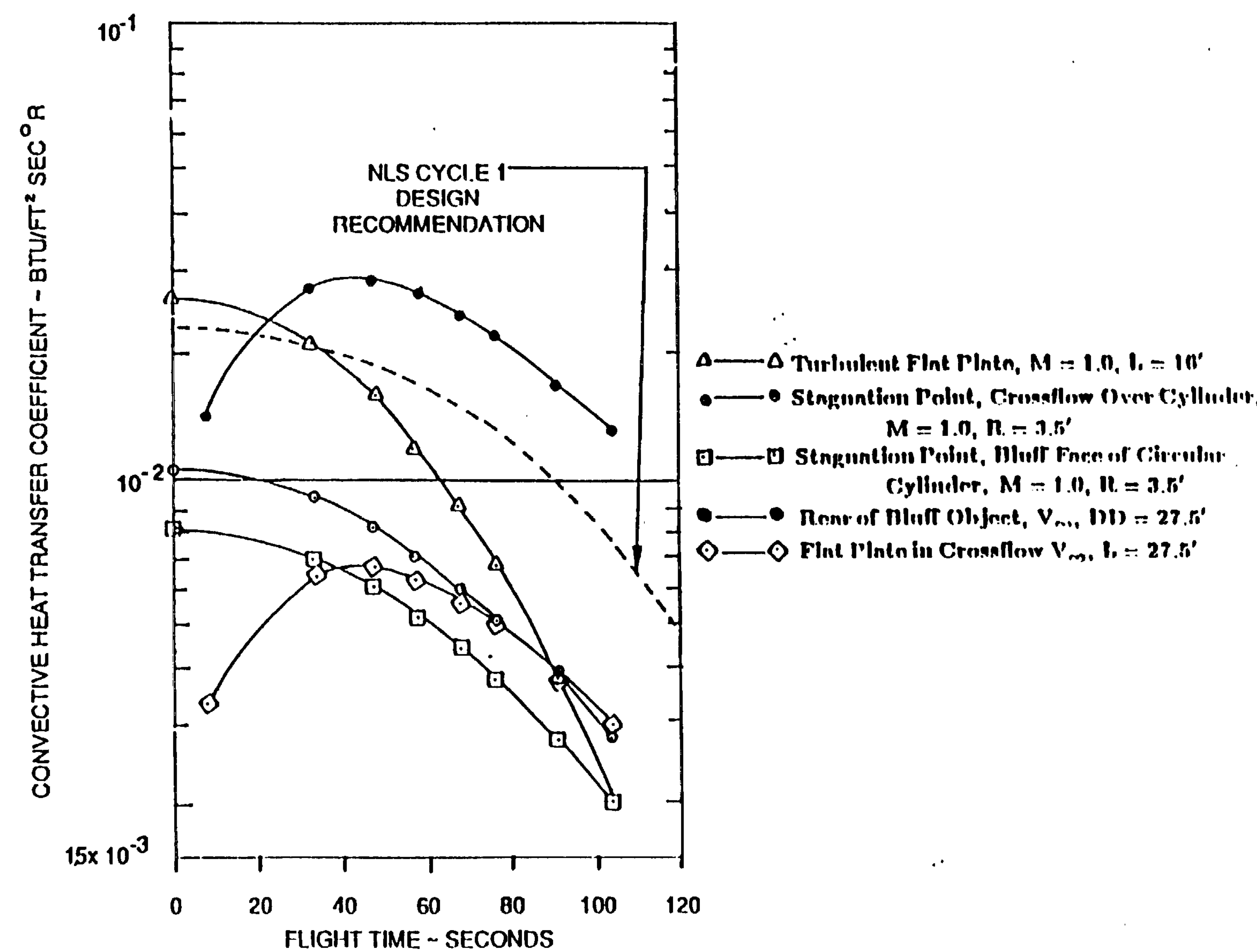
NLS - ESTIMATED CONVECTIVE BASE HEATING WITH TURBINE EXHAUST BASE BURNING



NLS - CONVECTIVE HEAT TRANSFER COEFFICIENT ESTIMATES FOR CORE BASE REGION



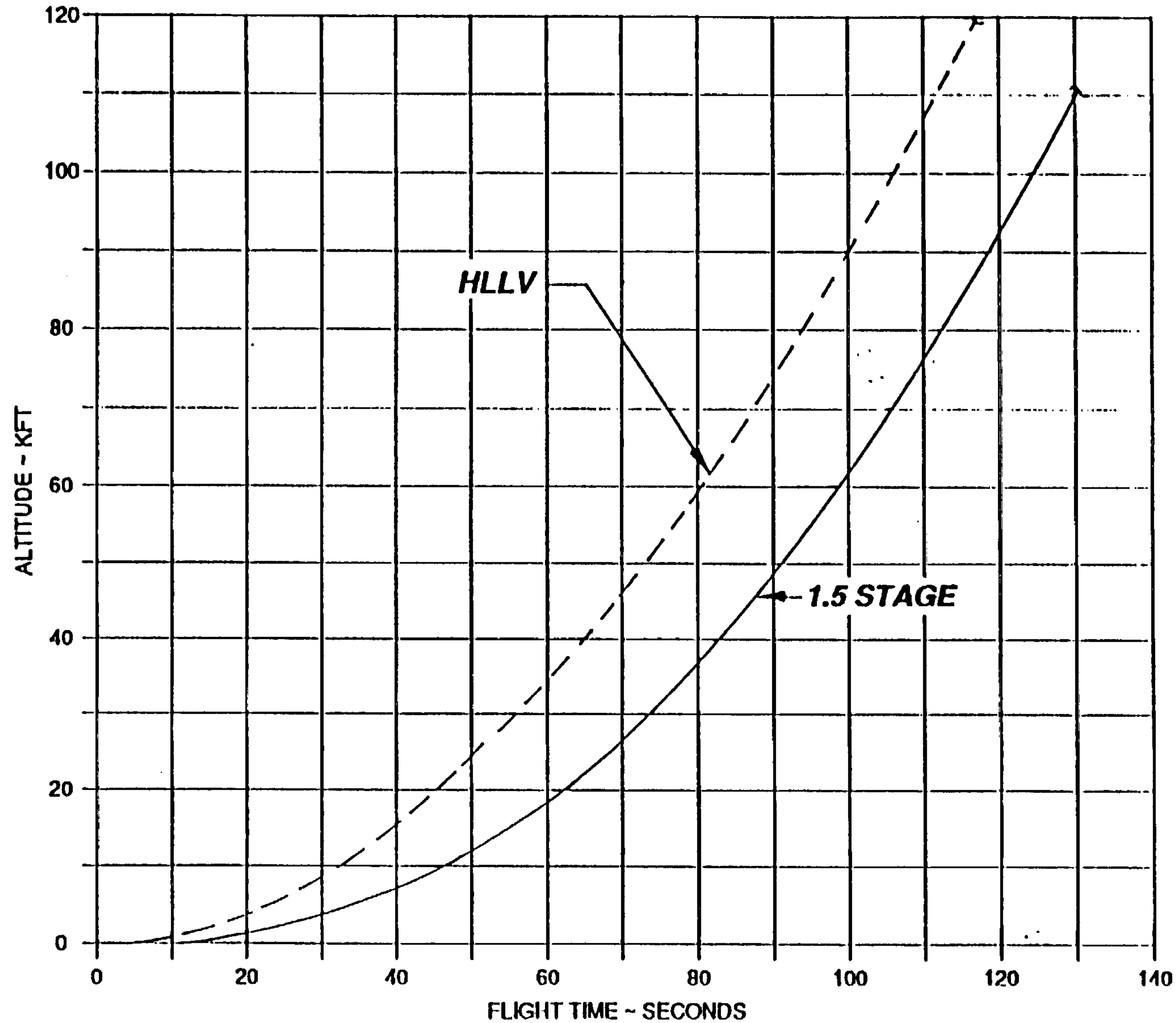
NLS - CONVECTIVE HEAT TRANSFER COEFFICIENT ESTIMATES FOR CORE BASE REGION



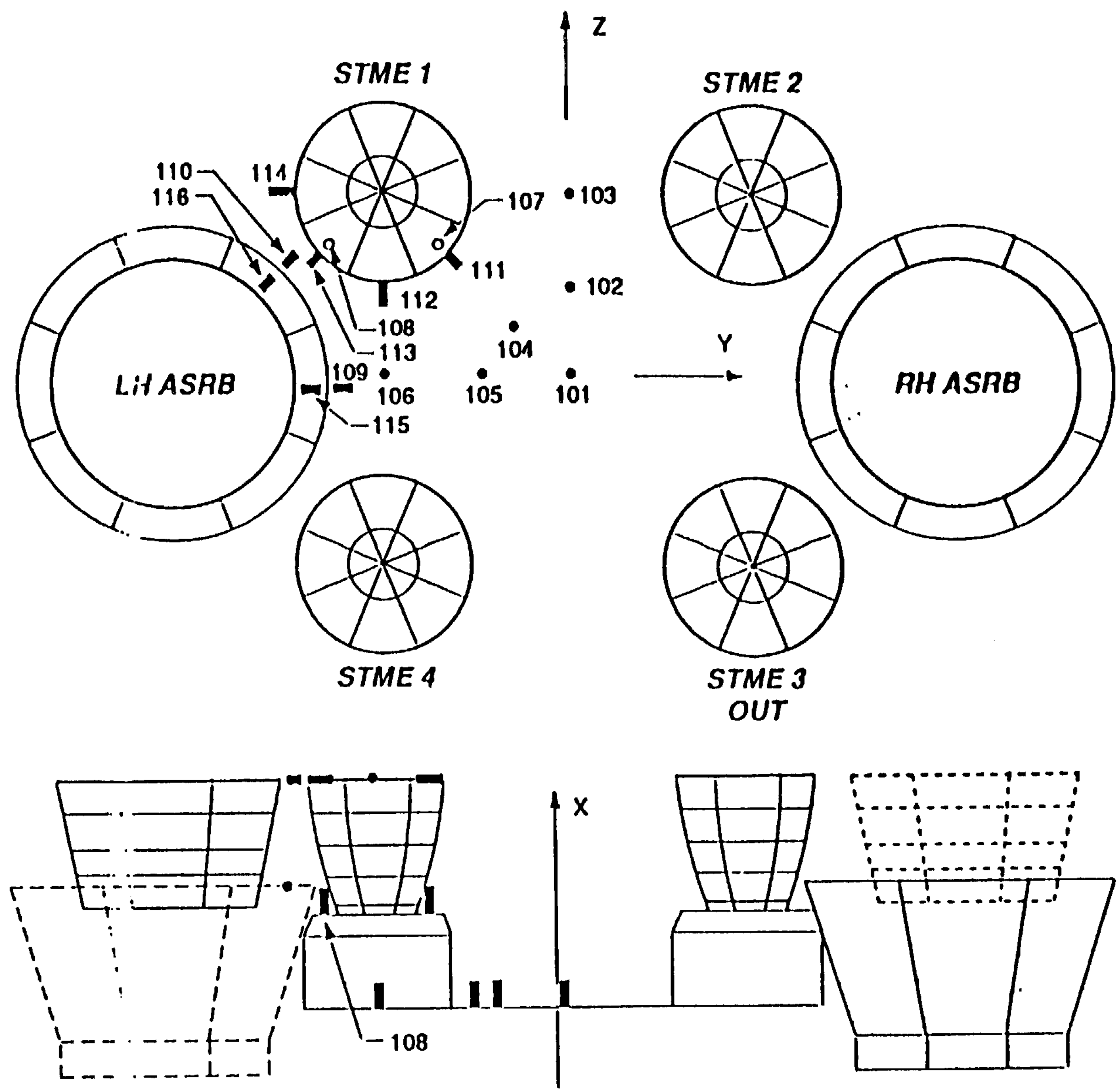
RESULTS OF BASE BURNING ANALYSIS

- Complex NLS base flowfields can recirculate low energy STME nozzle exhaust into base region at any altitude.
- Low energy plume boundary gases near nozzle lip will contain significant quantity of unburned H_2 and H_2O with current STME turbine exhaust disposal scheme.
- Burning of recirculate H_2 with air in base can occur from sea level to approximately 120,000 feet.
- Base gas temperatures as a result of H_2 burning can approach 4000° R at low altitudes.
- Convective heat transfer coefficients on the order of 2×10^{-2} BTU/ft^2 sec°R are feasible in the base at typical low altitude densities and turbulence levels.
- Convective heating rates as high as 80 BTU/ft^2 sec (cold wall) are possible.

NLS BASE HEATING TRAJECTORY - ALTITUDE vs TIME



HLLV BODY POINTS SELECTED FOR BASE HEATING ANALYSIS



NLS RADIATION ENVIRONMENTS AT SEA LEVEL

HLLV RADIATION ENVIRONMENTS

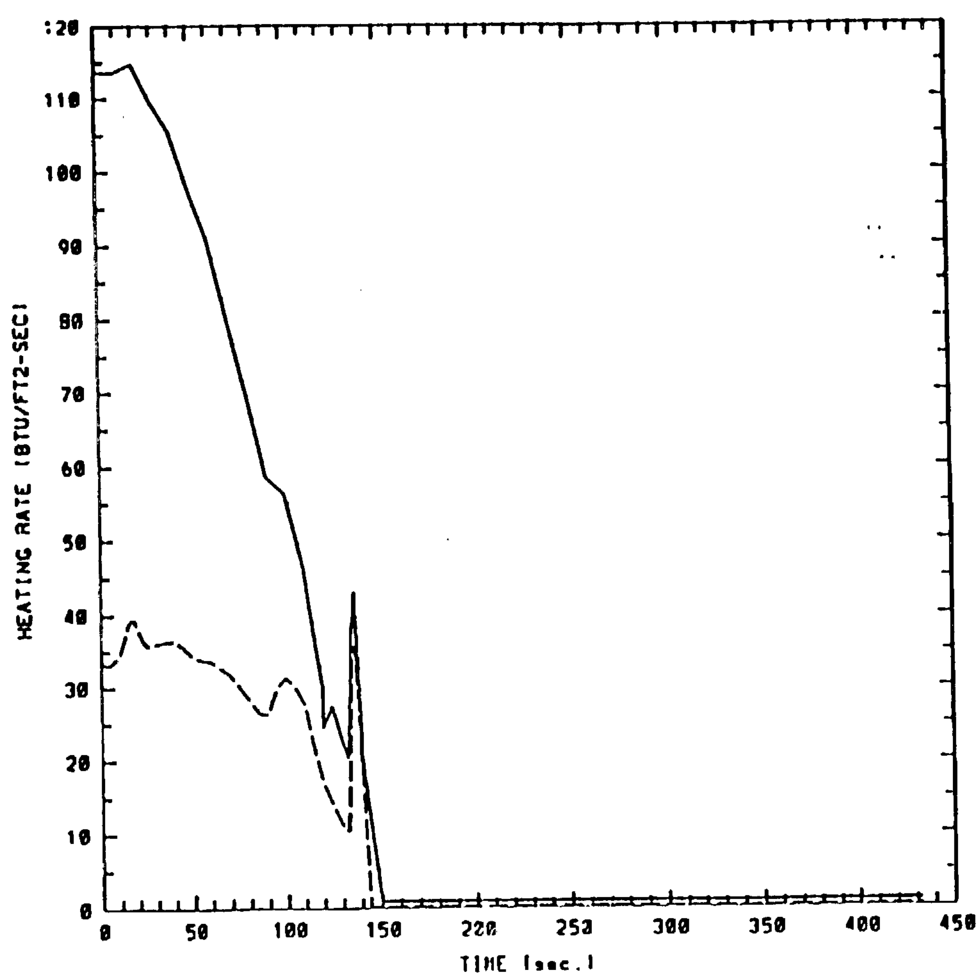
TIME (SEC)	HEATING RATE (BTU/FT SEC) FOR POINTS LISTED															
	101	102	103	104	105	106	107	108	109	110	111	112	113	114	115	116
0.0	26.52	25.56	20.51	25.48	25.65	21.24	23.69	21.08	7.15	2.27	17.27	20.83	33.39	17.59	19.28	15.26

1.5 STAGE RADIATION ENVIRONMENTS

TIME (SEC)	HEATING RATE (BTU/FT SEC) FOR POINTS LISTED											
	201	202	203	204	205	206	207	208	209	210	211	212
0.0	4.98	8.30	11.27	5.18	9.82	5.90	12.36	13.48	6.72	12.73	17.57	10.51

HLLV BASE HEATING ENVIRONMENTS AT B.P. 113

Radiation and Total Base Heating — HLLV STME Nozzle Exit Body Point 113



NLS CYCLE 1 ENVIRONMENT CONCLUSIONS

HLLV:

- Radiation dominated by ASRB plume radiation
- Predicted radiation from base gas burning is negligible
- Maximum radiation rate of 39.4 BTU/ft^{2}sec predicted for STME nozzle exit
- Convective heating resulting from turbine exhaust burning dominates the total environment during the first 120 seconds of ascent
- Maximum convective rate of 80.3 BTU/ft^{2}sec is predicted near sea level
- Core vehicle convection after ASRB separation is minimal

1.5 STAGE:

- Convective trajectory used becaue of difficulty in dealing with time mismatch; minor effect of radiation compared with convection
- Small increase caused by throttling (lower plume expansion)
- Altitude and magnitude of radiation from base burning very approximate
  Maximum radiation rate of 17.6 predicted for STME nozzle exit
- Convective heating due to turbine exhaust burning will dominate the environment during the first 120 seconds of ascent
- Maximum convective rate of 80.3 is predicted near sea level

IMPLICATIONS OF PROPOSED STME DESIGN CHANGES

- Upgrading current STME design to 650K has small impact (approx. 5 to 10% increase) on Cycle 1 environments.

- If STME remains G.G. cycle engine:
  1) Variations in nozzle disposal schemes have little impact on current conservative base burning analysis approach and resulting environments.
  2) Outboard ducts change base burning potential - but have not been analyzed.

- Regenerative cooled dual combustion engine similar to SSME would effectively eliminate low altitude base burning

NLS BASE HEATING/BASE BURNING

VERY NEAR TERM FOLLOW-ON ANALYSIS PLAN

- Revisit flight data
  - Saturn 1 Block I (SA-1 through SA-4)
  - Saturn V/S-1C Stage (501 through 505)

- Reconstruct heat transfer coefficient envelopes from flight data

- Adjust envelopes to NLS flight conditions

- Reevaluate choice of "original" Saturn I flight deduced heat transfer coefficient for NLS design environment

- Propose new coefficient, if indicated

NLS

CONVECTIVE BASE HEATING INVESTIGATION

SATURN FLIGHT DATA REVIEW

MAY 19, 1992

Presented by:
ROBERT L. BENDER
REMTECH inc

OUTLINE

- Problem Definition
- Objectives of Saturn flight data review
- Overview of Saturn flight programs
- Overview of Saturn base heating flight measurements
- Task 1 Results - h_c derived from Saturn flight data
- Task 2 - T_{Gas} from Saturn flight data trends
- Application of results to NLS 1.5 Stage vehicle
- Conclusions

PROBLEM DEFINITION

Convective Heating Rate $q_c = h_c(T_c - T_W)$

- NLS Cycle 1 Ascent Convective Base Heating Predictions, MSFC Memo ED33 (15-92), assume hydrogen in the STME turbine exhaust will be recirculated and combusted in the NLS base region from near sea level to approximately 120,000 feet altitude.

- Combustion of the hydrogen with air is assumed to occur at stoichiometric mixture ratios at base pressure conditions which are approximately equal to ambient pressure - resulting in base gas temperatures ranging from 4200^o R near sea level to 3700^o R when combustion ceases.

- Local flowfields in the NLS base region cannot be computed with accuracy and experimental data for the NLS does not exist at this time; therefore, the convective heat transfer coefficient was approximated based upon Saturn I flight deduced trends adjusted to the NLS trajectory.

- Because the convective environment is relatively severe and results in significant TPS penalties for the base region including the STMEs, a detailed review of applicable Saturn flight data during early ascent when base burning may have occurred was recommended to verify or improve the Cycle 1 environment.

NLS CONVECTIVE BASE HEATING
NEAR TERM* ANALYSIS OBJECTIVES

*Re-directed effort beginning March 1, 1992, following publication of Cycle 1 environments

- ***SATURN FLIGHT DATA REVIEW AND ANALYSIS***
  - *Task 1:*
    - Reduce all Saturn flight data during first 100,000 feet of flight to convective heat transfer coefficient and compare with Cycle 1 design recommendation.

$$h_c = \frac{Q_{Total} - Q_{Radioation}}{T_{Gas} - T_{Wall\ of\ Total\ Cal}}$$

  - *Task 2:*
    - Use Saturn V, S-1C Stage flight measured gas temperatures to deduce air/turbine exhaust mixtures in base.
    - Adjust these mixture ratios to NLS 1.5 Stage STME turbine exhaust conditions, then compute NLS base gas temperature.
    - Compare with Cycle 1 design gas temperature recommendation.

NLS BASE HEATING ENVIRONMENTS
NEAR TERM ANALYSIS SCHEDULE

REMTECH

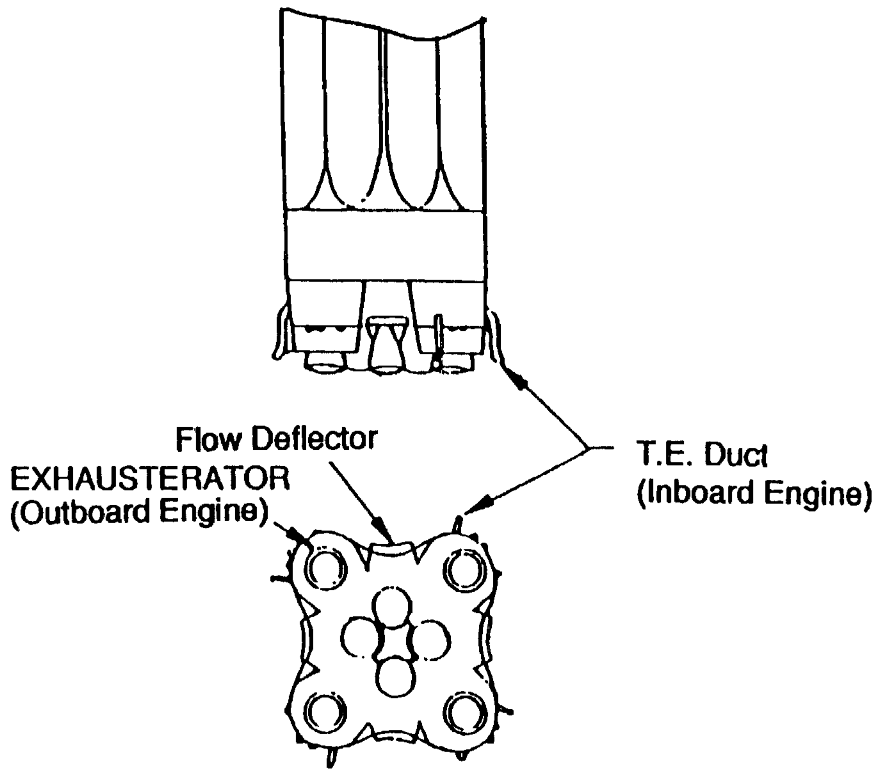
EVENT	CY 1992 Feb	Mar	Apr	May	Jun	Jul	Aug	Sep
• ***CYCLE 1 DESIGN ENVIRONMENT***								
∘ ED 33 (15-92) Memo Out	▲ 2/2							
∘ NLS Chief Engineer Briefing	▲ 2/8							
∘ NLS Contractor Briefing	2/20 ▲							
• ***FOLLOW-ON ANALYSIS***								
∘ Saturn Data Review								
- Begin		▲ 3/2						
- Task 1-h_c From Flight								
- Task 2 - T_{GAS} From Saturn V								
- Review Briefings								
- Preliminary - ED33/TFG			4/17 ▲					
- ED Lab Working Group				▲ 5/13				
- First Formal - ED33/TFG-JW				5/19 ▲				
- Advisory Committee					▲ 6/11			
- NLS Project/Contractor					6/26 ▲			
- Publish "New" Q_c Environment						▲ 7/3		

OVERVIEW OF SATURN FLIGHT PROGRAMS

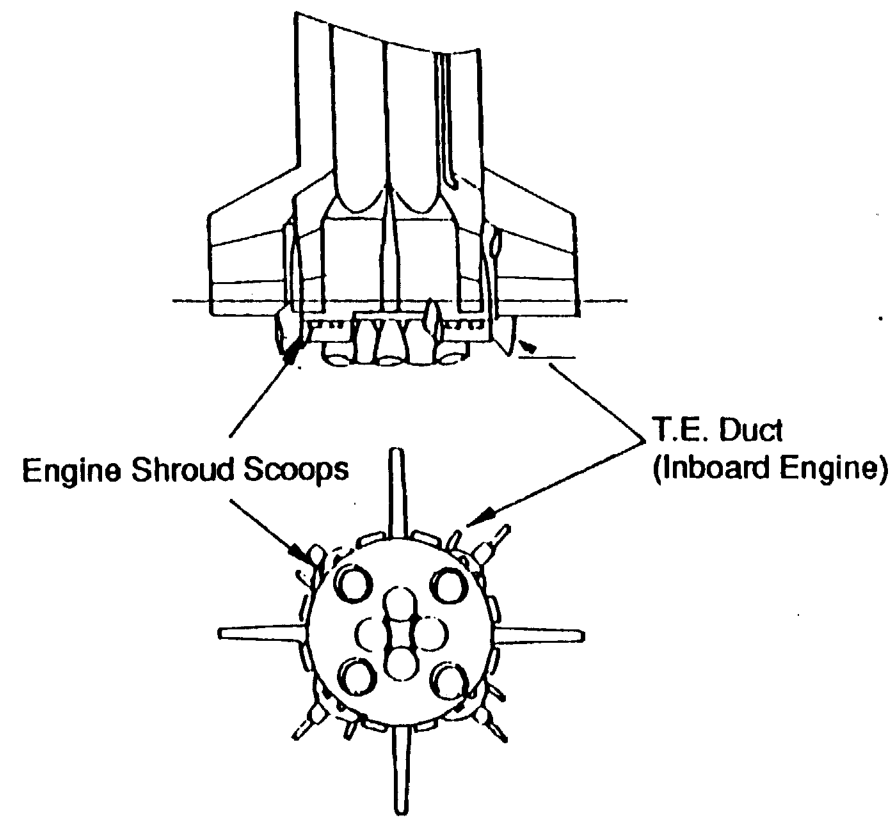
SUMMARY OF SATURN FLIGHT VEHICLES

VEHICLE	FIRST STAGE DESIGNATION	FLIGHT DESIGNATION	FIRST STAGE ENGINE DESCRIPTION				
			DESIG	PROP	THRUST	NO.	TYPE
Saturn I Block I	S-I	4 Flights SA-I to SA-4	H-1	LOX/RP-1	165K	8	Gas Generator Cycle
Saturn I Block II	S-I	6 Flights SA-5 to SA-10	H-1	LOX/RP-1	188K	8	Gas Generator Cycle
Saturn IB	S-IB	4 Flights AS-201 to AS-204	H-1	LOX/RP-1	200K	8	Gas Generator Cycle
Saturn V	S-IC	12 Flights AS-501 to AS-512	F-1	LOX/RP-1	1.53M	5	Gas Generator Cycle

SATURN FLIGHT VEHICLE BASE CONFIGURATIONS

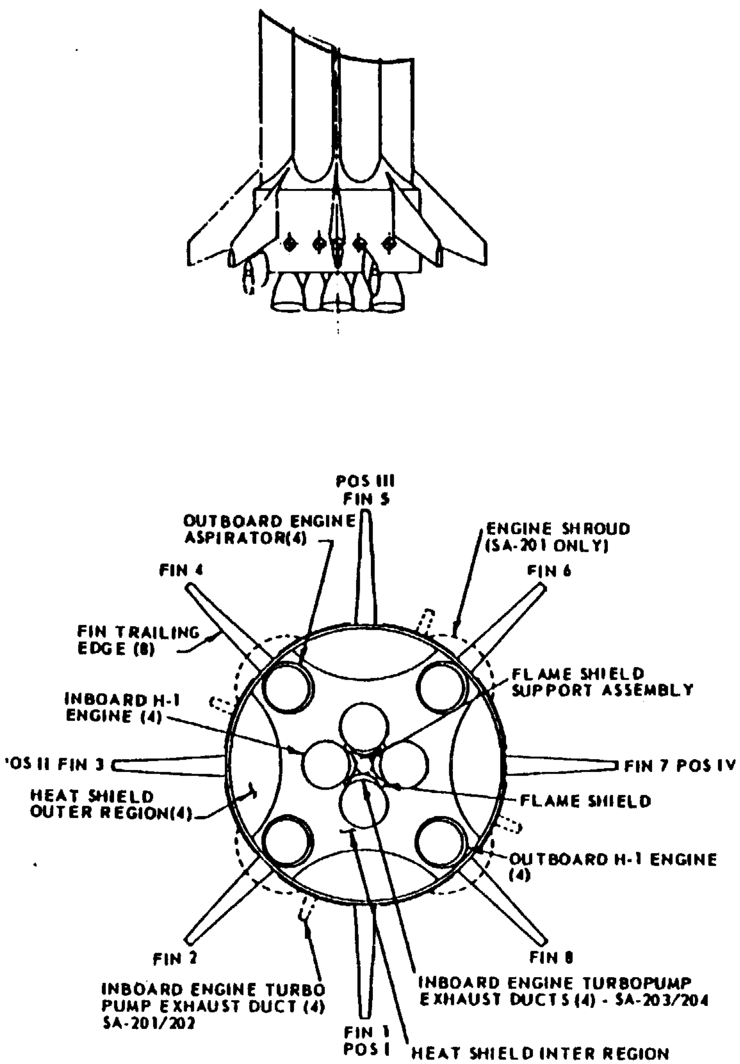


SATURN I, BLOCK I

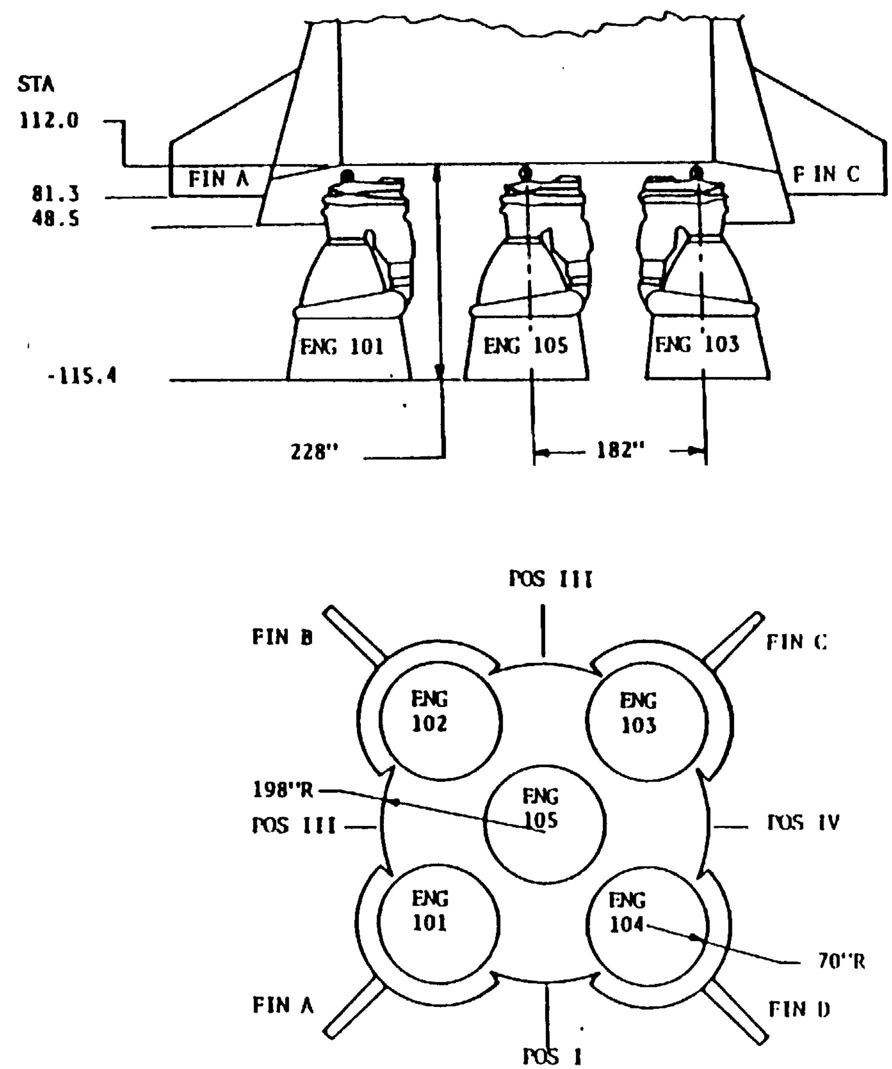


SATURN I, BLOCK II

SATURN FLIGHT VEHICLE BASE CONFIGURATIONS



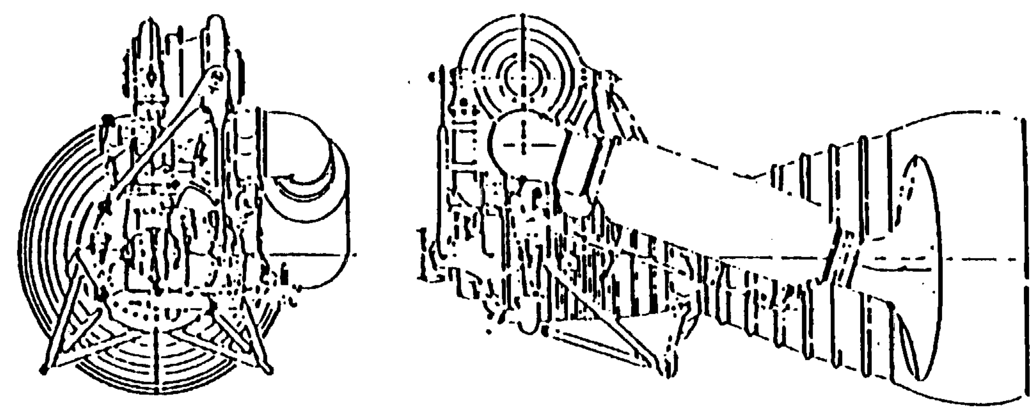
SATURN IB



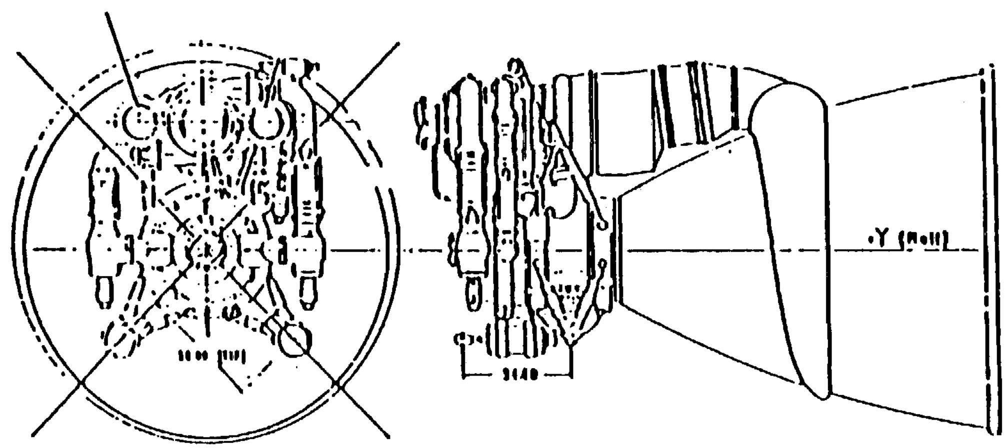
SATURN V

TURBINE EXHAUST DISPOSAL INSIDE NOZZLE

REMTECH



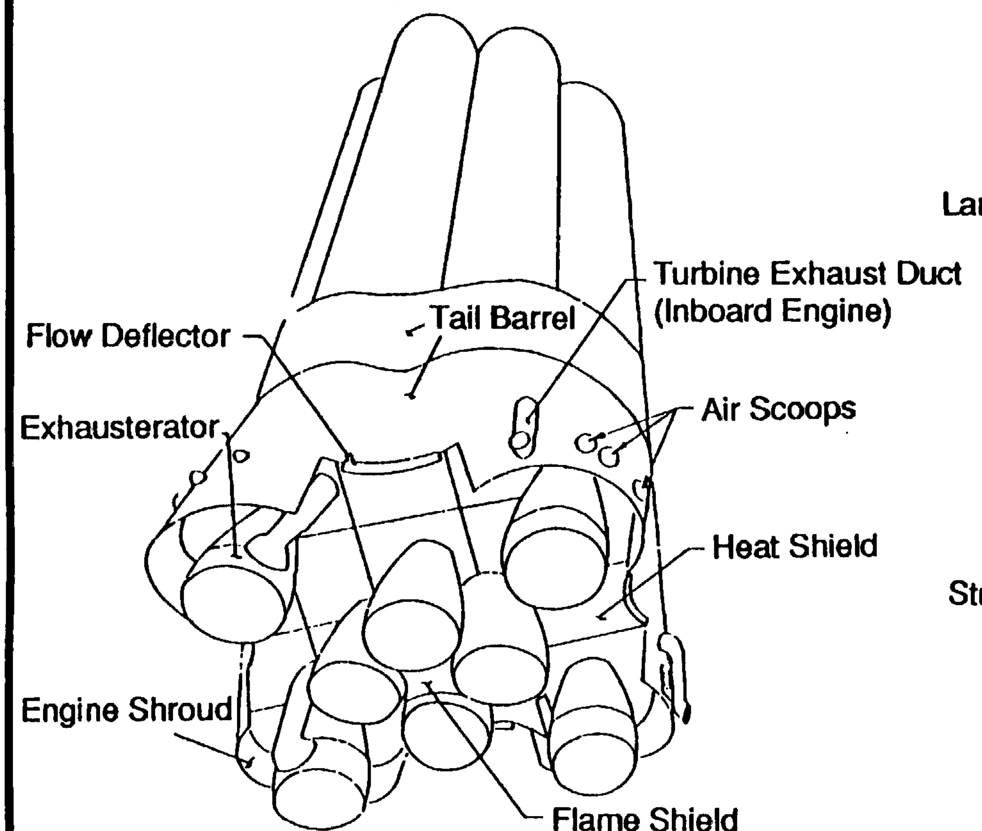
SATURN I AND IB BOOSTERS - OUTBOARD H-1 ENGINE



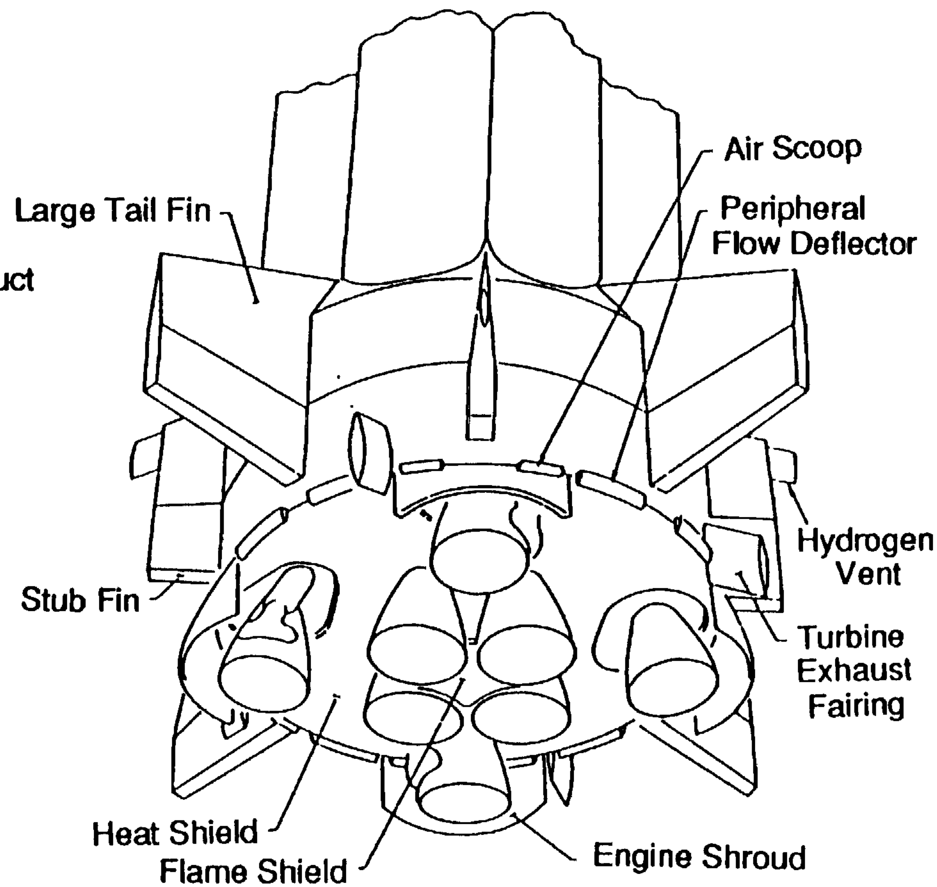
SATURN V/S-1C STAGE - F-1 ENGINE

SCOOPS, FLOW DEFLECTORS, AND TURBINE EXHAUST DUCTS

REMTECH

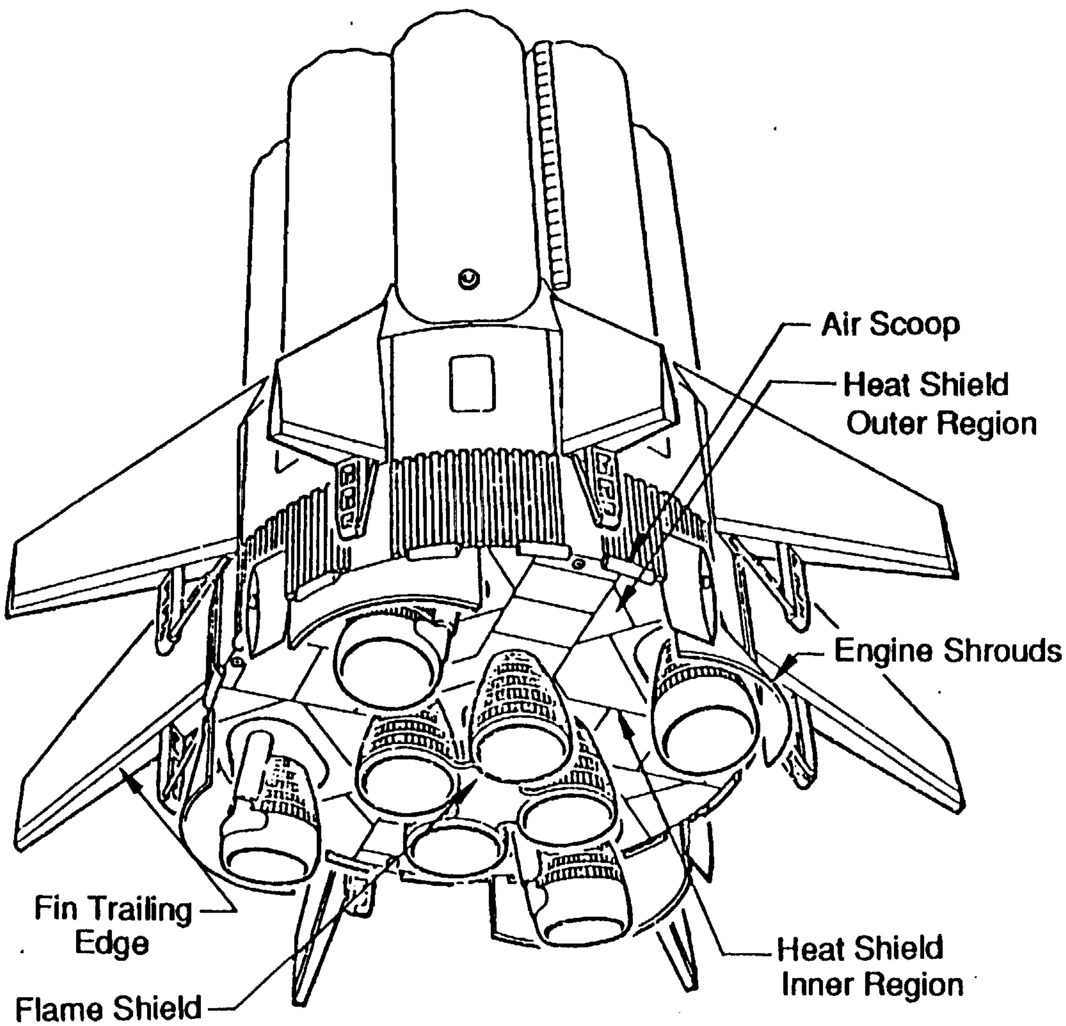


SATURN I, BLOCK I

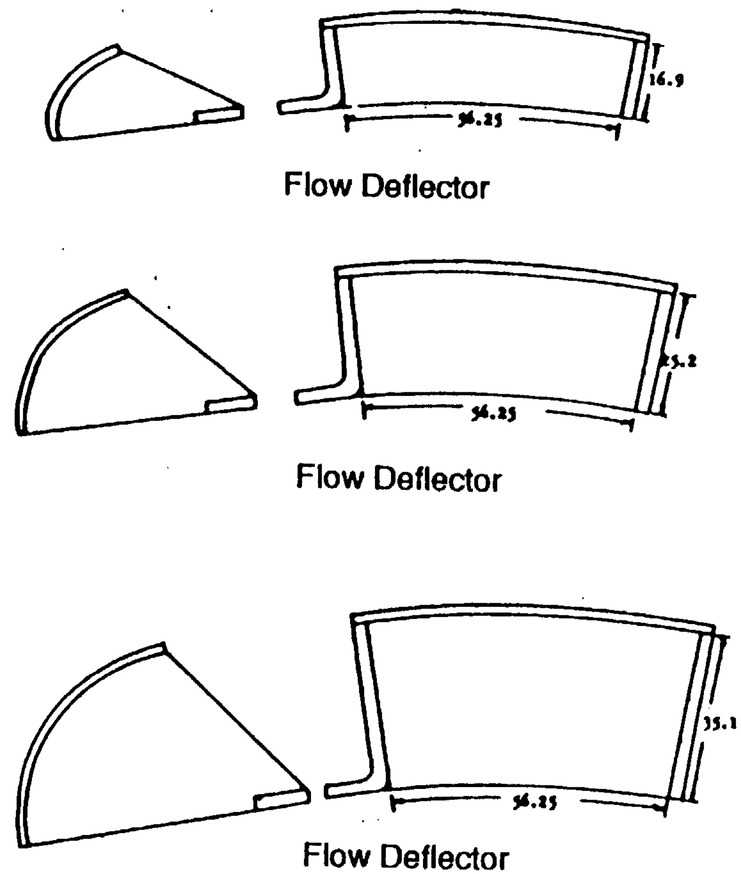


SATURN I, BLOCK II

SCOOPS, FLOW DEFLECTORS, AND TURBINE EXHAUST DUCTS

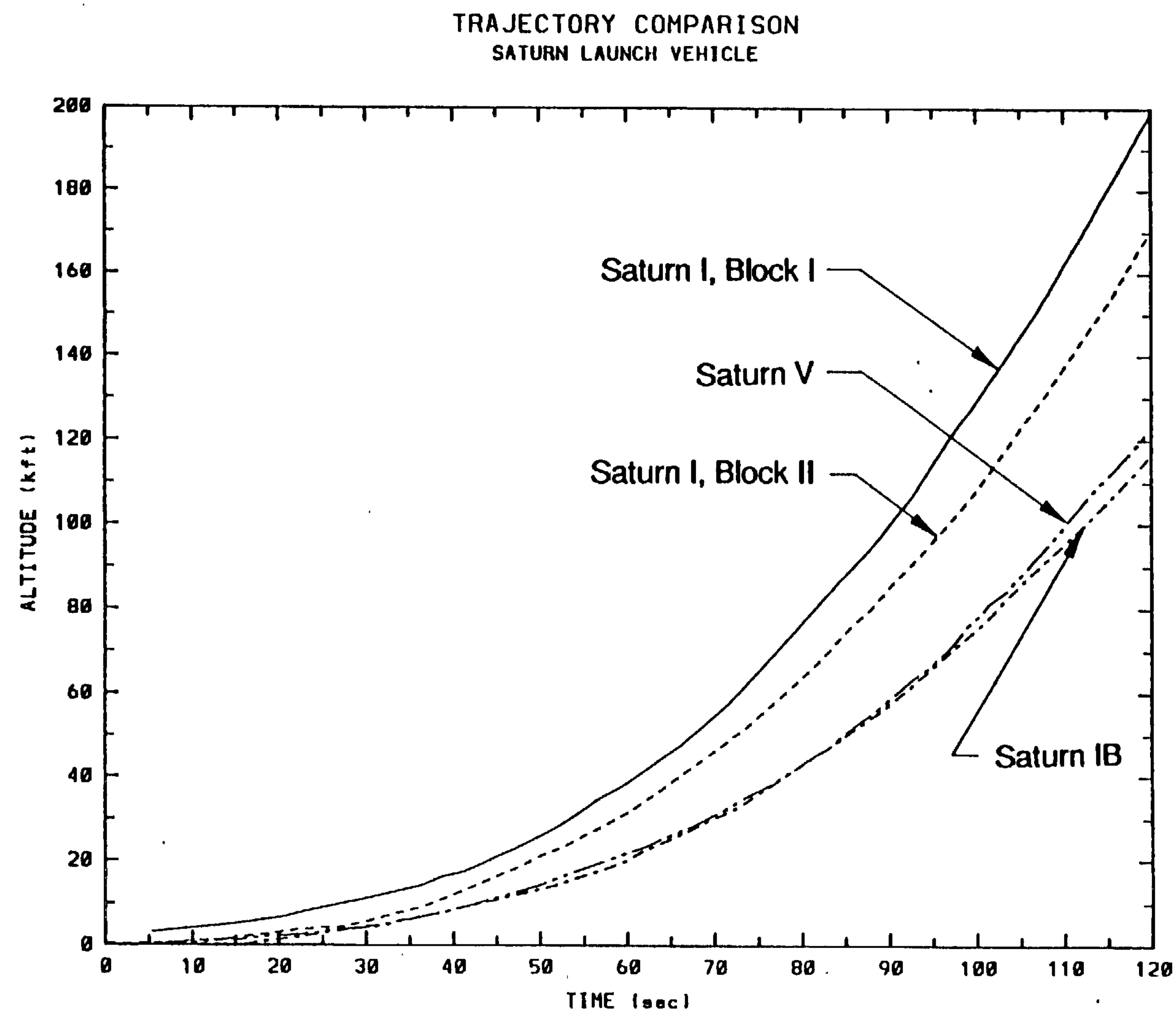


SATURN IB

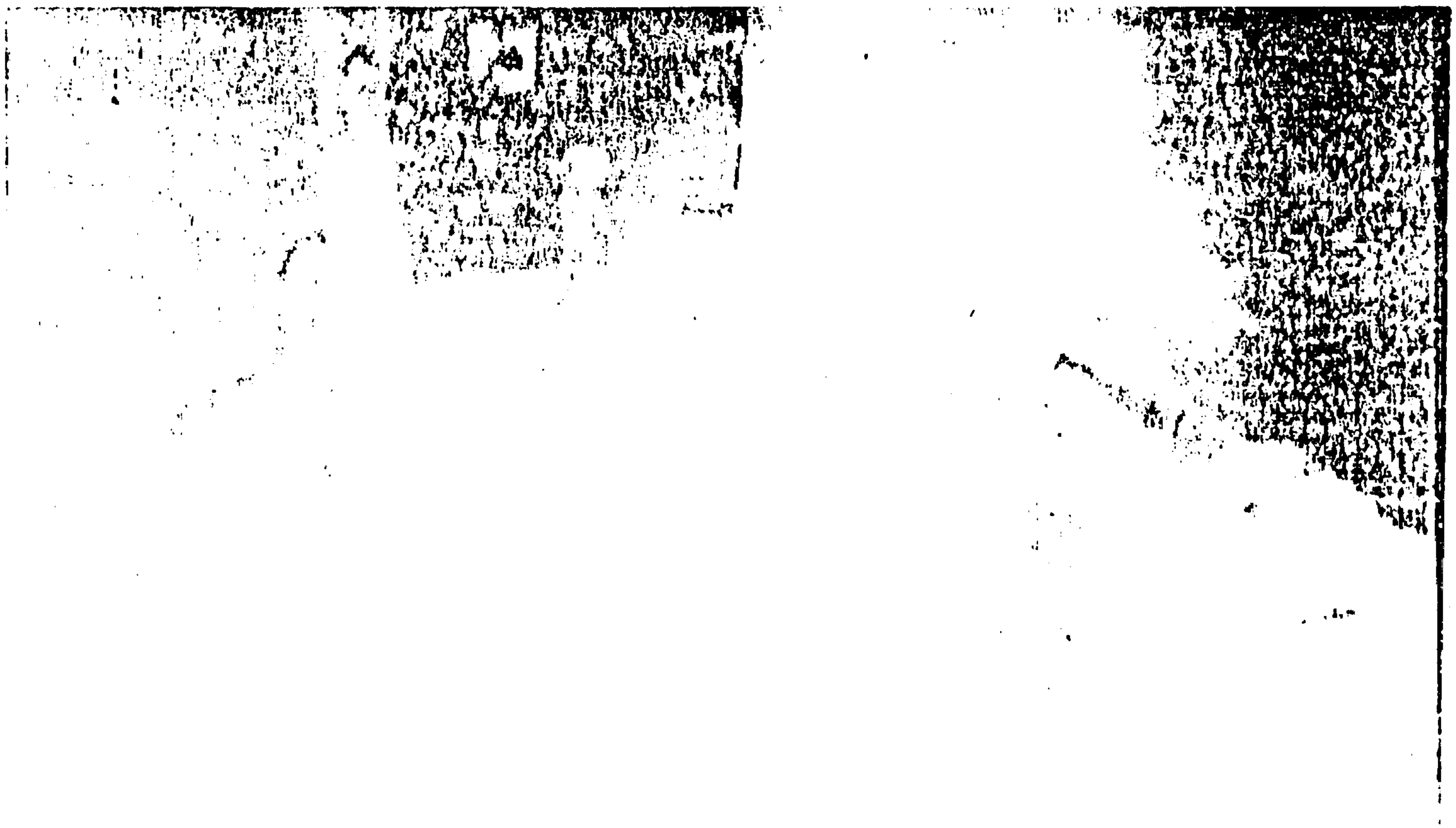


SATURN V FLOW DEFLECTORS

TYPICAL SATURN FLIGHT VEHICLE TRAJECTORIES



SATURN I BASE FLOWS NEAR LIFTOFF



SATURN V BASE FLOWS NEAR LIFT-OFF

REMTECH

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OF POOR QUALITY

REMTECH

OVERVIEW OF SATURN BASE HEATING INSTRUMENTATION

SATURN FLIGHT BASE HEATING INSTRUMENTATION SUMMARY

REMTECH

VEHICLE	FLIGHT	BASE HEAT SHIELD			ENGINES		
		TOT CAL	RAD	GTP	TOT CAL	RAD	GTP
SAT I BK I	SA-1	3	2	5	0	0	0
	SA-2	3	2	5	0	0	0
	SA-4	3	2	5	0	0	0
	SA-4	3	2	5	0	0	0
SAT I BKII	SA-5	5	5	7	0	0	0
	SA-6	5	5	7	0	0	0
	SA-7	5	5	7	0	0	0
	SA-8	5	5	7	0	0	0
	SA-9	5	5	7	0	0	0
	SA-10	5	5	7	0	0	0
SAT IB	AS-201	3	2	2	3	0	0
	AS-202	3	2	2	0	0	0
	AS-203	1	0	0	3	0	0
	AS-204	3	3	3	5	0	1
SAT V	AS-501	5	3	6	12	3	7
	AS-502	5	3	6	12	3	7
	AS-503	5	3	6	12	3	7
	AS-504	4	3	6	12	3	1
	AS-505	4	3	6	12	3	1
	AS-506	2	0	2	0	0	0
	AS-507	2	0	2	0	0	0
	AS-508	2	0	2	0	0	0
	AS-509	2	0	2	0	0	0
	AS-510	2	0	2	0	0	0
	AS-511	2	0	2	0	0	0
	AS-512	2	0	2	0	0	0

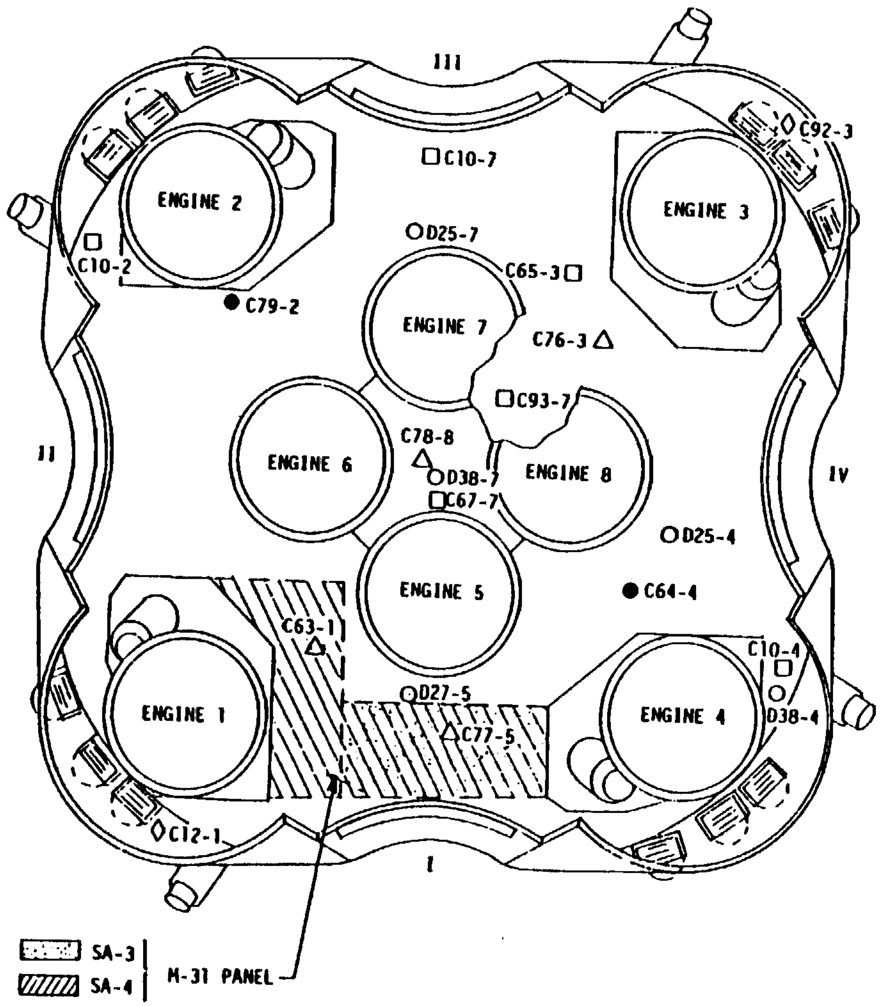
SATURN FLIGHT BASE HEATING
INSTRUMENTATION GENERAL DESCRIPTIONS

REMTECH

INSTRUMENT	COMMENTS
Total Calorimeters	•Slug type, Saturn I, Bk I & Bk II •Membrane, Saturn IB & Saturn V •Typical ranges: 0 - 40, 0 - 100 BFS — Sat . I & IB 0 - 40, 0 - 60, 0 - 100 BFS — Sat. V
Radiometers	•Slug type with sapphire window, Saturn I, Bk I & Bk II •Membrane with sapphire window, Saturn IB •Membrane with sapphire window and nitrogen purge, Sat. V •Typical ranges: 0 - 40 BFS, Sat. I & IB H.S. 0 - 100, BFS, Sat. I & 1B F.S. 0 - 20 BFS, Sat. V H.S. 0 - 60, 0 - 100 BFS, Sat. V Engine
Gas Temperature Probes	•Unshielded, single & double shield T/C - Sat. I, Bk I & II •Exposed T/C with Guard Ring, Sat. IB •Double Shielded (Platinum) T/C, Sat. V •Typical ranges: 0 - 1500, 0 - 1750, 0 - 2000°C Sat. I & IB 0 - 1750°C, Sat. V

SATURN BASE HEATING INSTRUMENTATION
TYPICAL PATTERNS

REMTECH



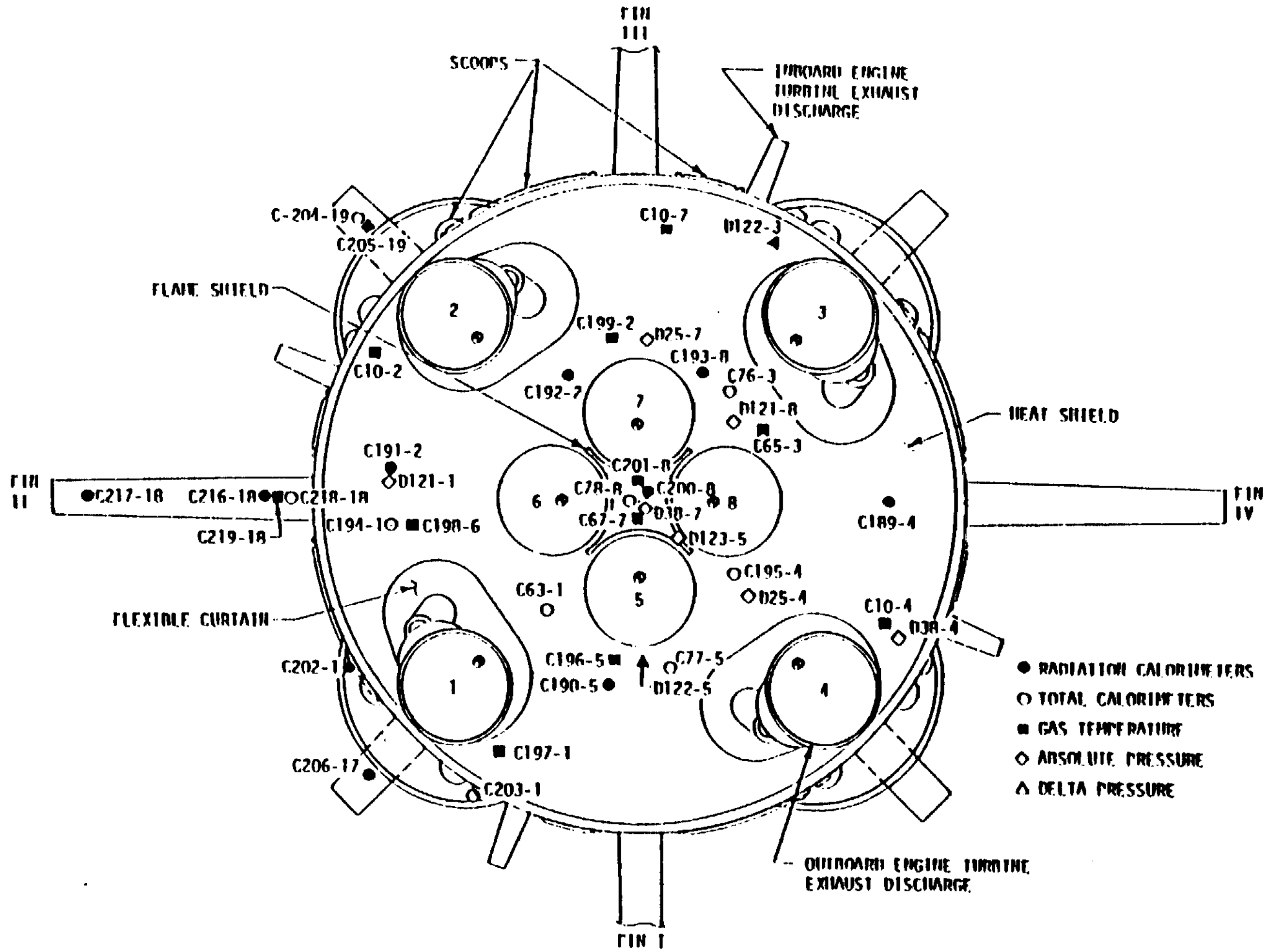
○ PRESSURE MEASUREMENT
● RADIATION CALORIMETER
△ TOTAL CALORIMETER
□ GAS TEMPERATURE
◇ SKIN TEMPERATURE

NOTE: MEASUREMENTS MOUNTED ON 9-in. STANDOFF

C63-1 (SA-3 ONLY)
C64-4
C65-3
C76-3
C77-5 (SA-1 AND SA-2 ONLY)
C79-2
C93-7

SATURN I, BLOCK I

SATURN BASE HEATING INSTRUMENTATION
TYPICAL PATTERNS



SATURN I, BLOCK II

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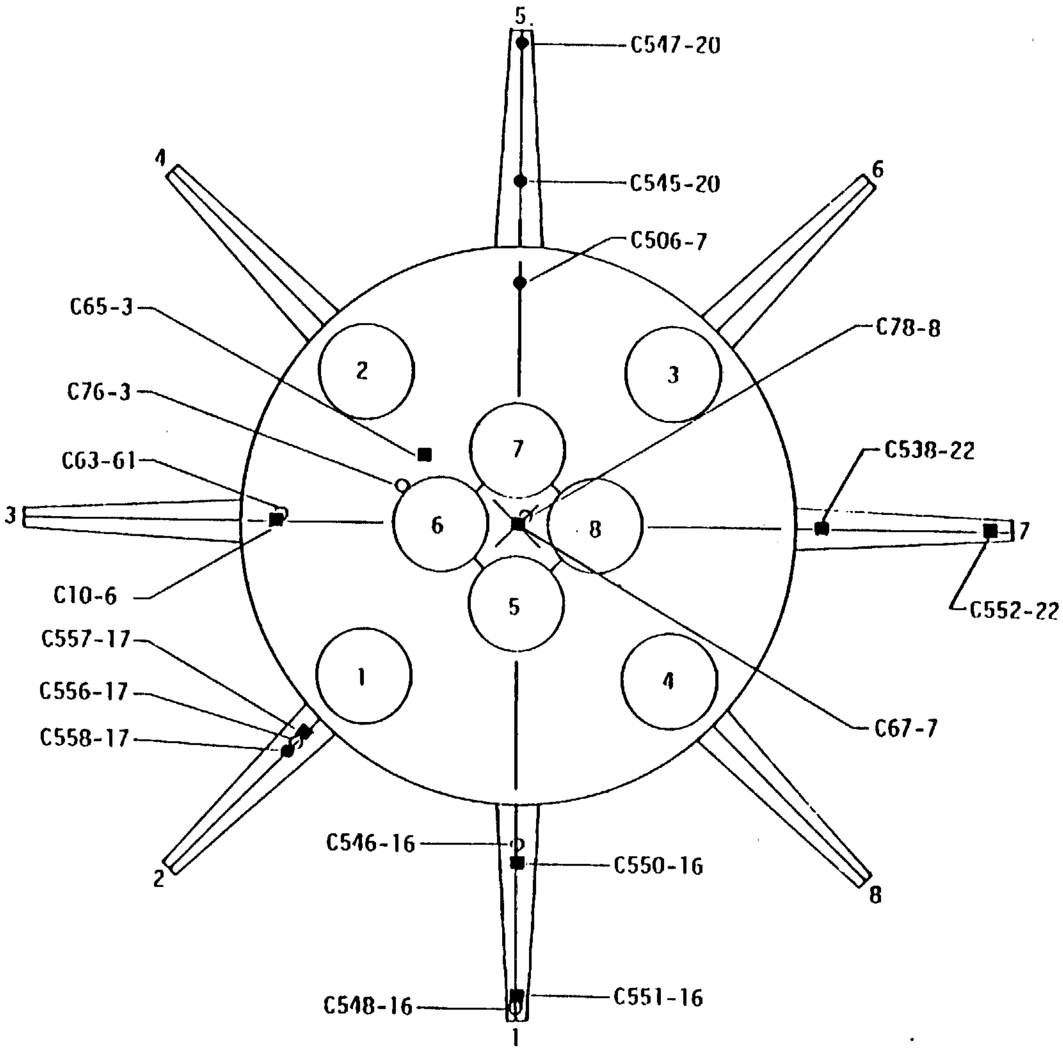
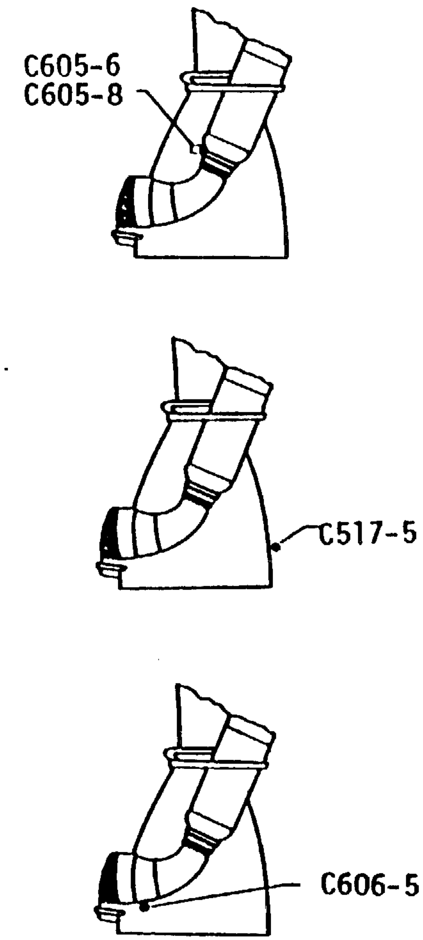
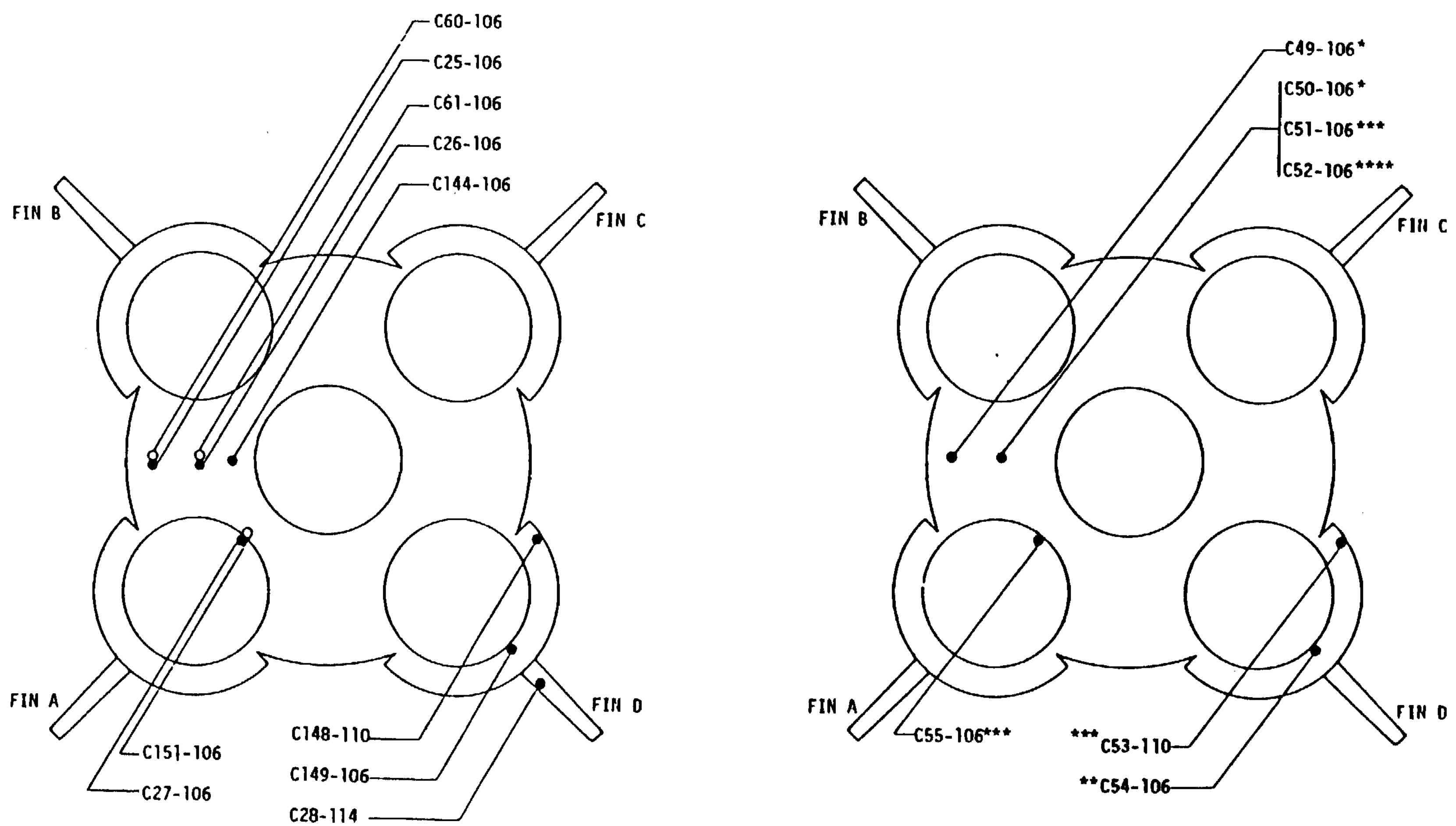
SATURN BASE HEATING INSTRUMENTATION
TYPICAL PATTERNS





SATURN IB

SATURN BASE HEATING INSTRUMENTATION
TYPICAL PATTERNS



● TOTAL CALORIMETER

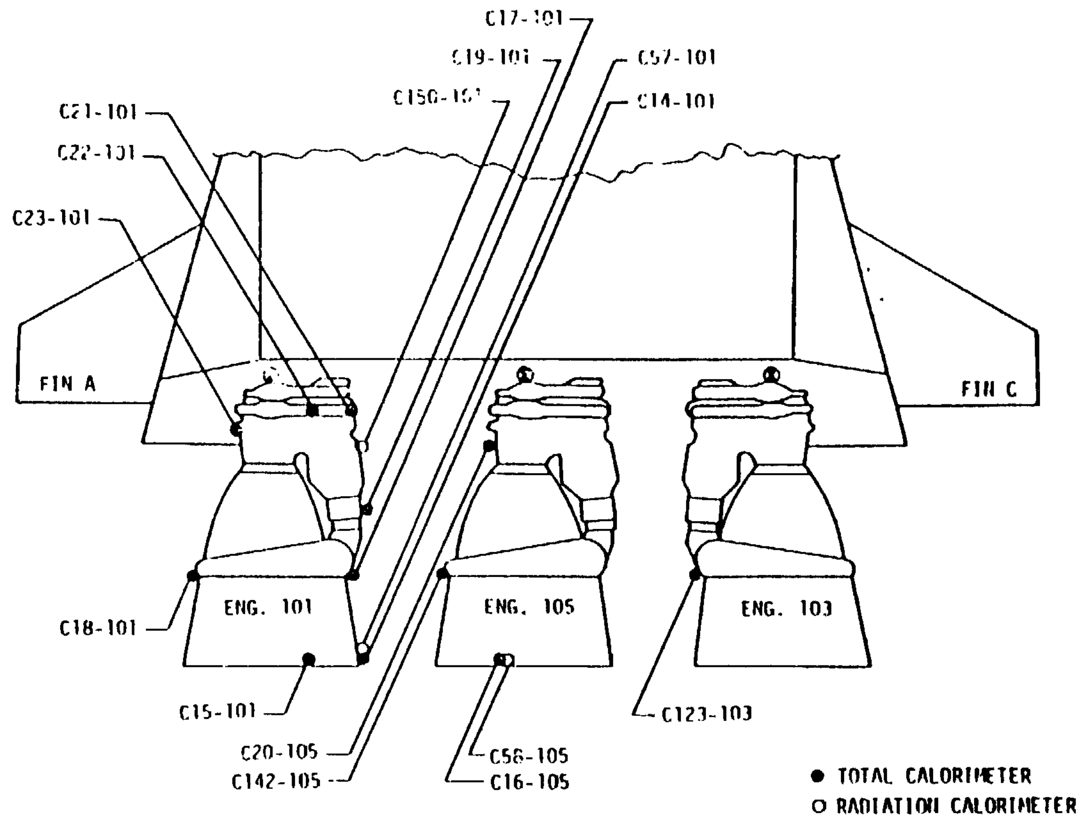
○ RADIATION CALORIMETER

● GAS TEMPERATURE PROBE

* - 0.25-in. OFF SURFACE
** - 0.50-in. OFF SURFACE
*** - 1.0-in. OFF SURFACE
**** - 2.5-in. OFF SURFACE

SATURN V S-1C STAGE

SATURN BASE HEATING INSTRUMENTATION
TYPICAL PATTERNS



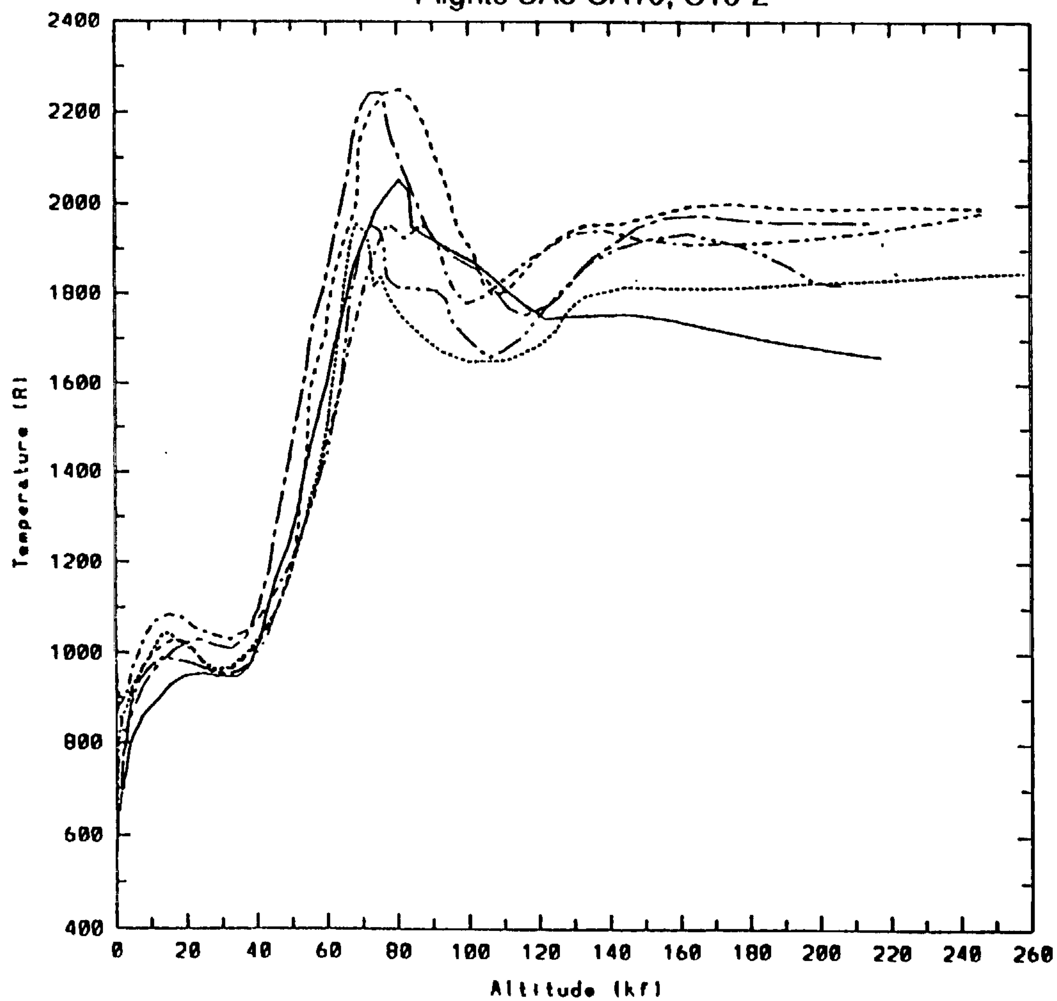
SATURN V S-1C STAGE F-1 ENGINES

SATURN FLIGHT BASE HEATING DATA REPEATABILITY

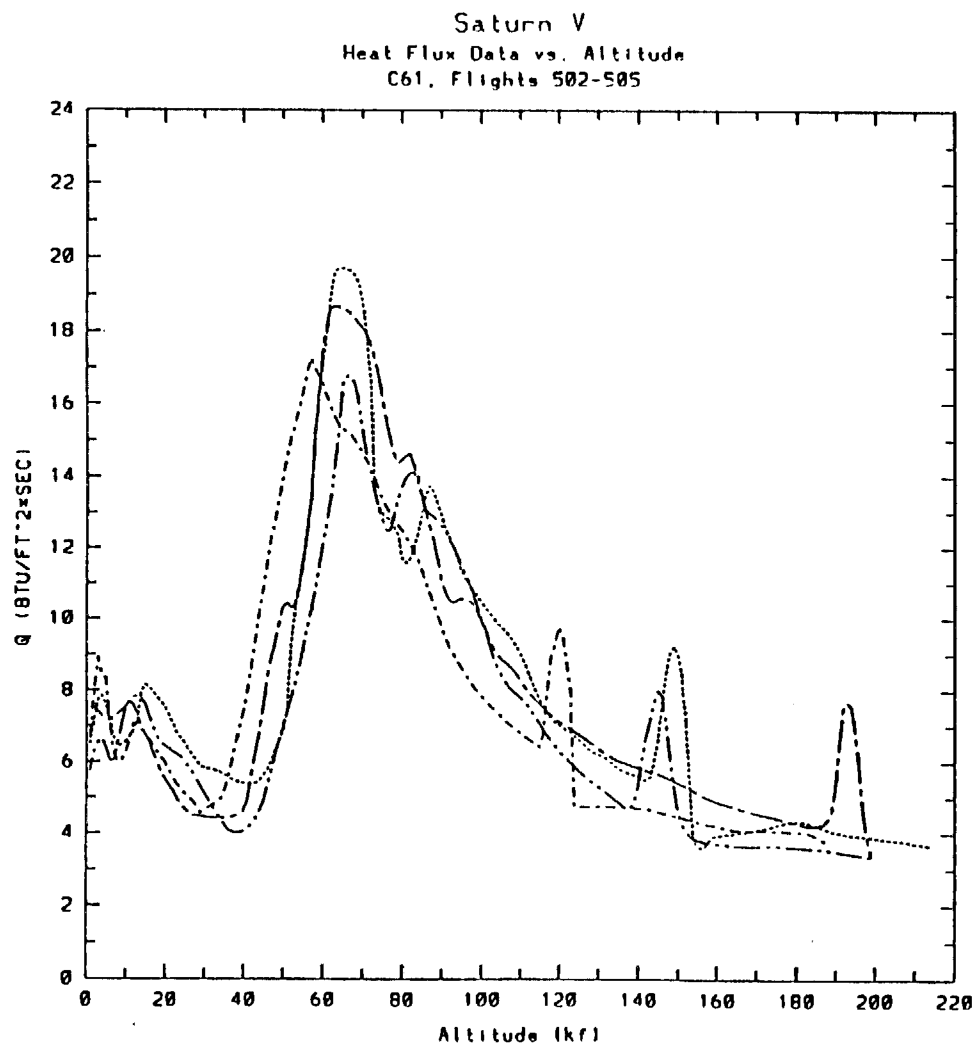
Saturn I Block II

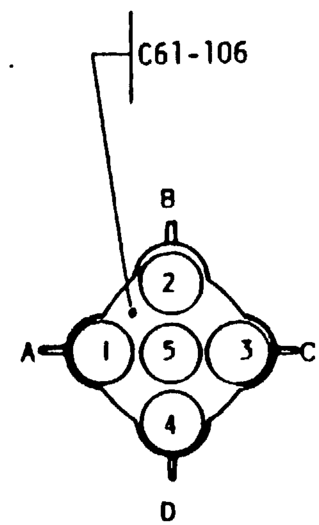
Gas Temperature Data vs. Altitude

Flights SA5-SA10, C10-2



SATURN FLIGHT BASE HEATING DATA REPEATABILITY





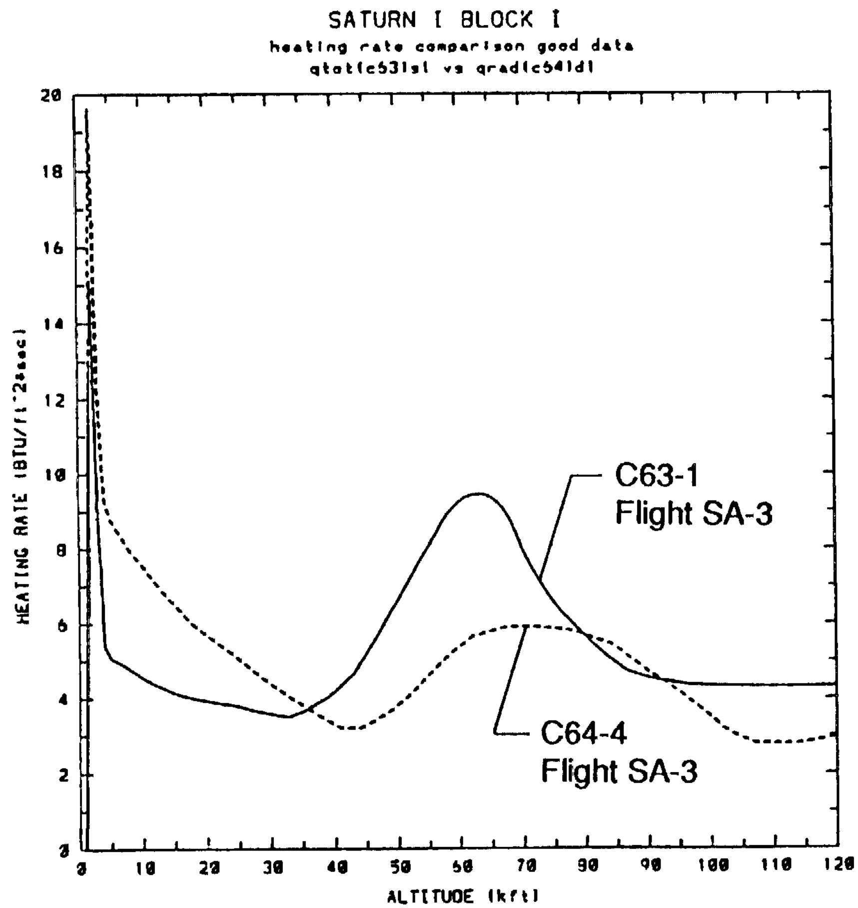
TASK 1 - h_c DERIVED FROM SATURN FLIGHT DATA

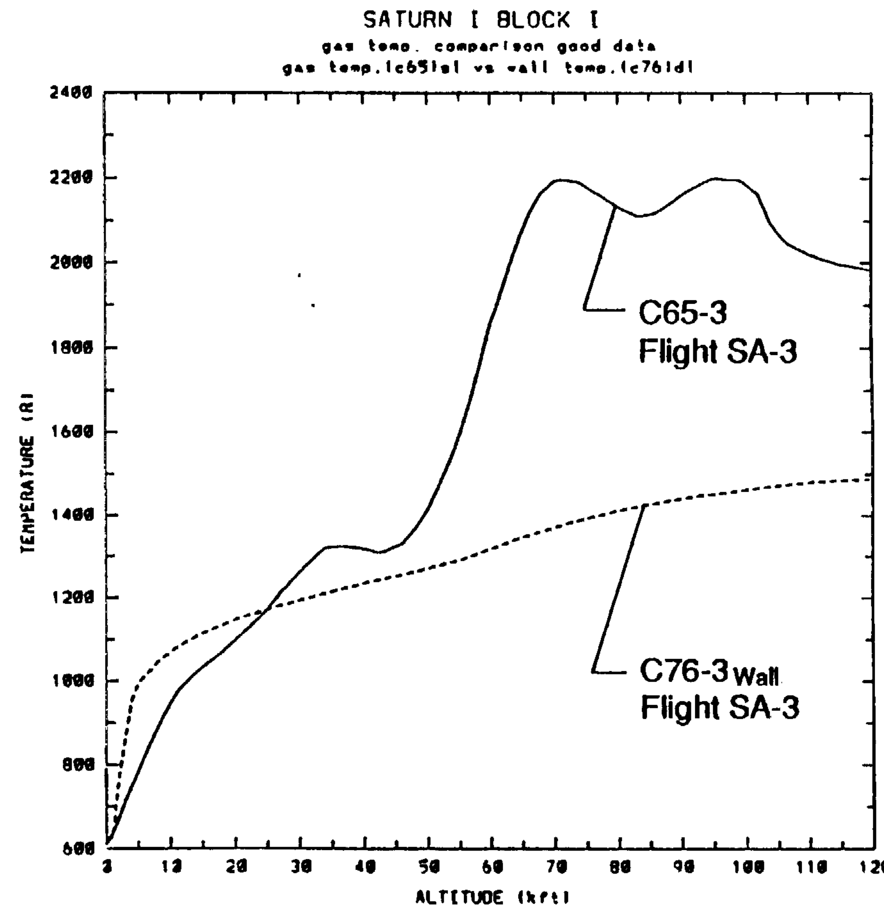
INSTRUMENT GROUPING FOR h_c REDUCTION

$$h_c = \frac{Q_{Tot} - Q_{Rad}}{T_{Gas} - T_{Wall}}$$

VEHICLE		INSTRUMENT			
		TOT CAL	RAD	GTP	T_{WALL}
SATURN I BK I HEAT SHIELD		C76-3 C63-1	C79-2 C64-4	C65-3 C65-3	C76-3(wall) Flight SA-4 C63-1(wall) FLight SA-4
SATURN I BK I HEAT SHIELD		C76-3 C63-1 C194-1 C194-1	C193-8 C190-5 C192-2 C189-4	C65-3 C196-5 C198-6 C10-4	C76-3 (wall) SA-4 C63-1 (wall) SA-4 C76-3 C76-3 (wall) SA-4
SATURN IB	Heat Shield	C611-3 C508-3	C609-3 C506-7	C610-3 C507-3	C233-106 AS-504 C233-106 AS-504
	H-1 Engine	—	—	—	—
SATURN V	Heat Shield	C25-106 C26-106 C27-106	C60-106 C61-106 C151-106	C49-106 C50-106 C55-106	C233-106 C233-106 C233-106
	F-1 Engine	C14-101 C14-101	C57-101 C57-101	C44-101 C56-105	C224-105 C234-106

SATURN I BLOCK I TYPICAL "GOOD" BASE HEATING DATA





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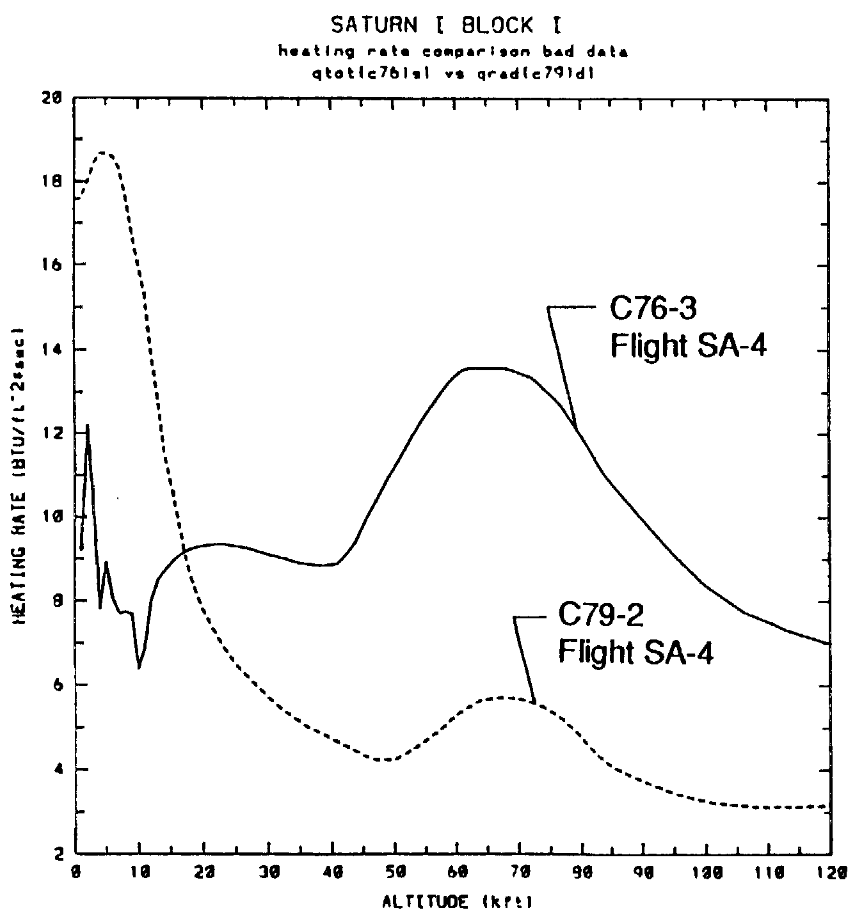
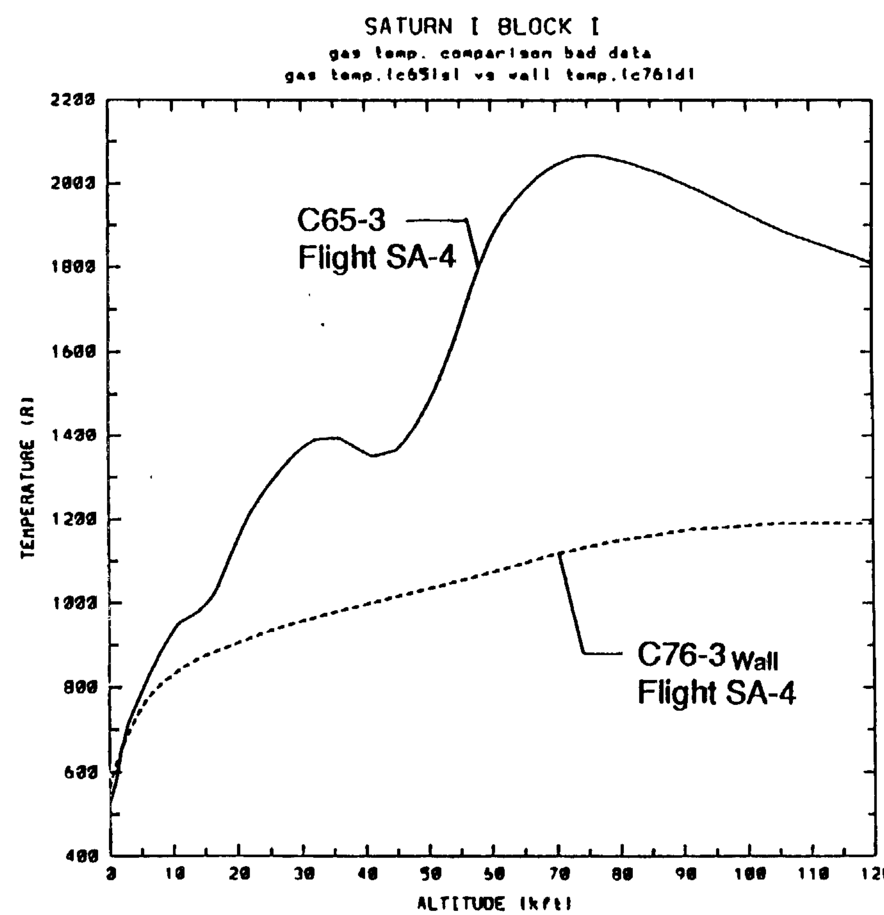
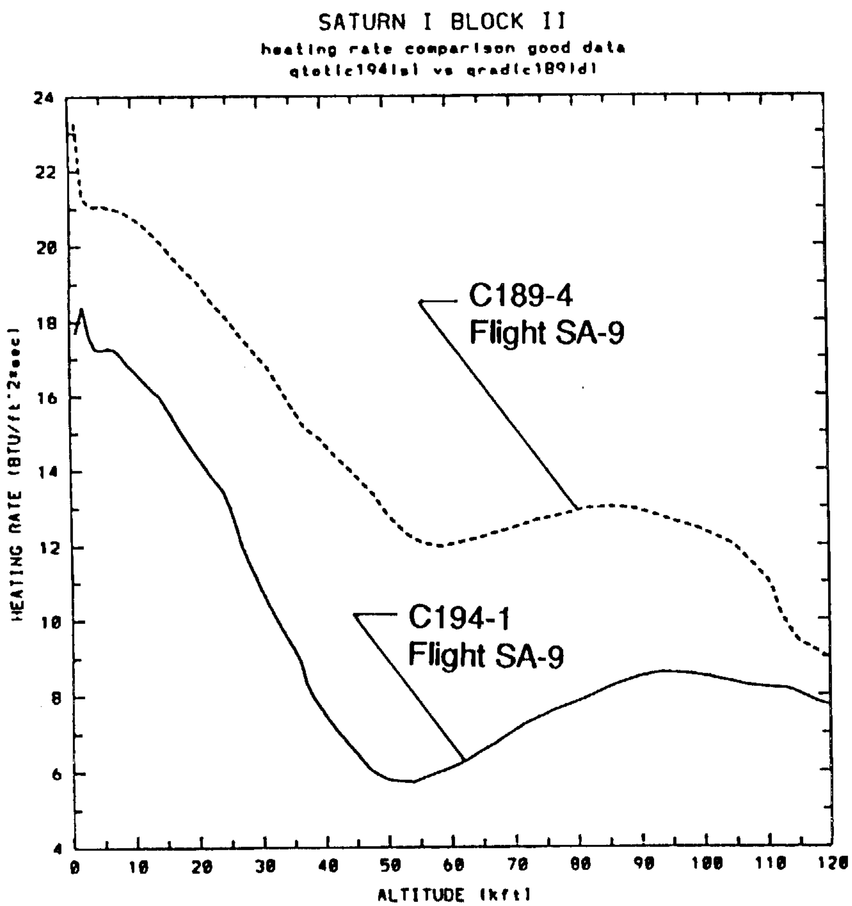
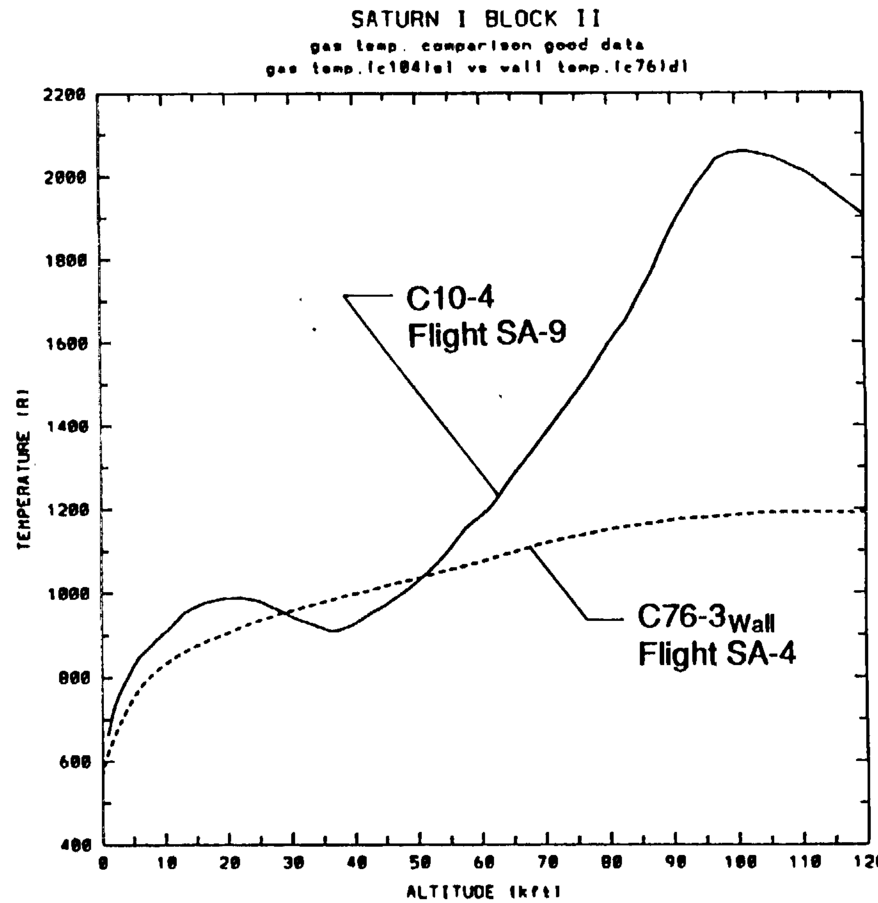


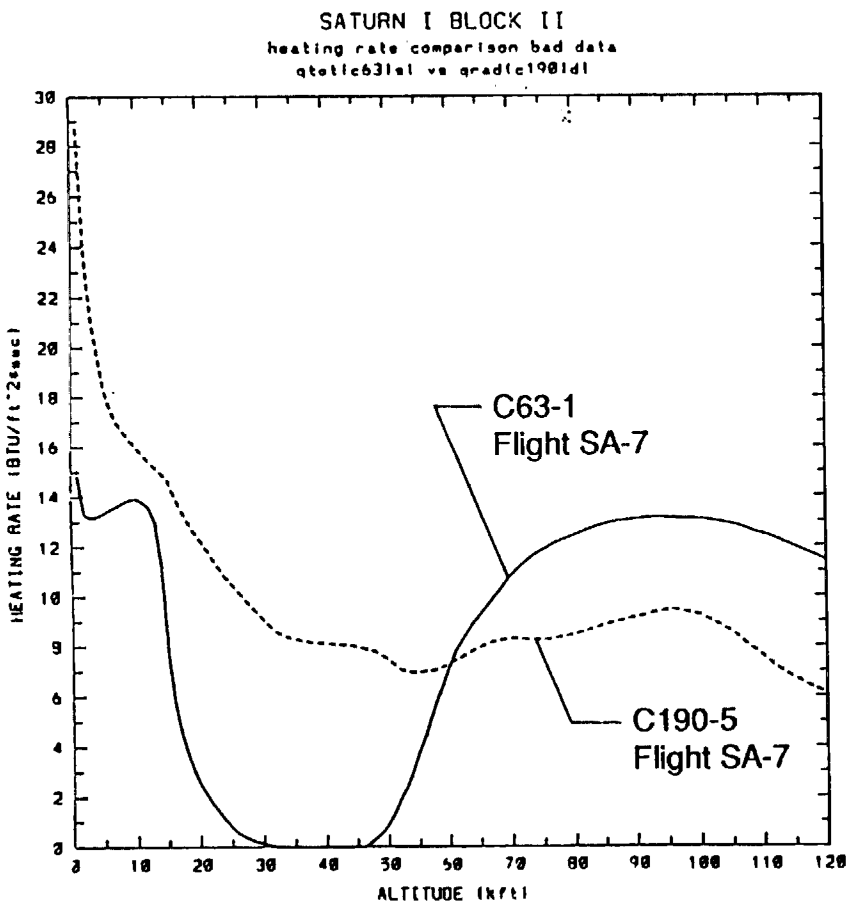


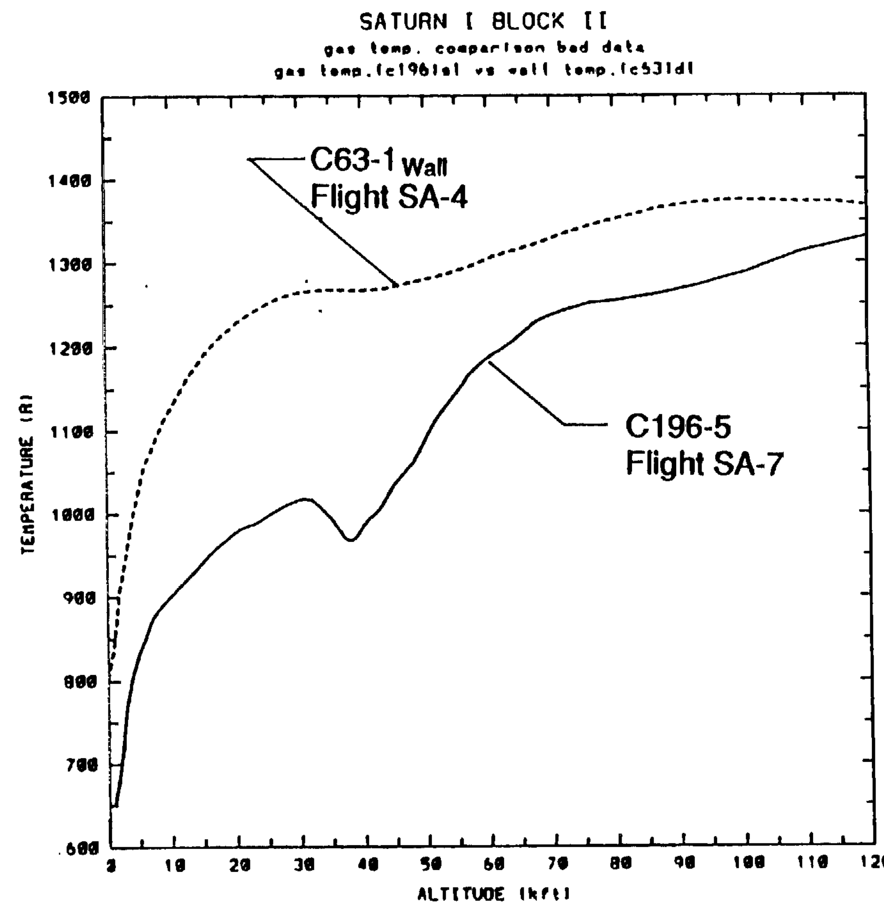
SATURN I BLOCK II TYPICAL "GOOD" BASE HEATING DATA



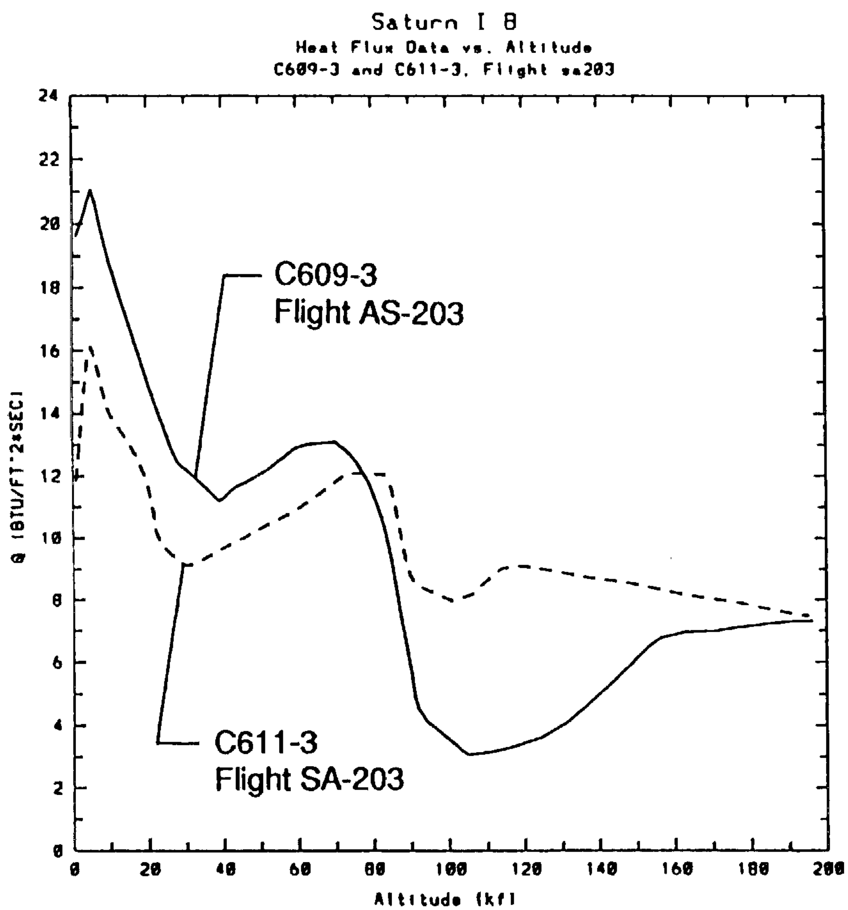


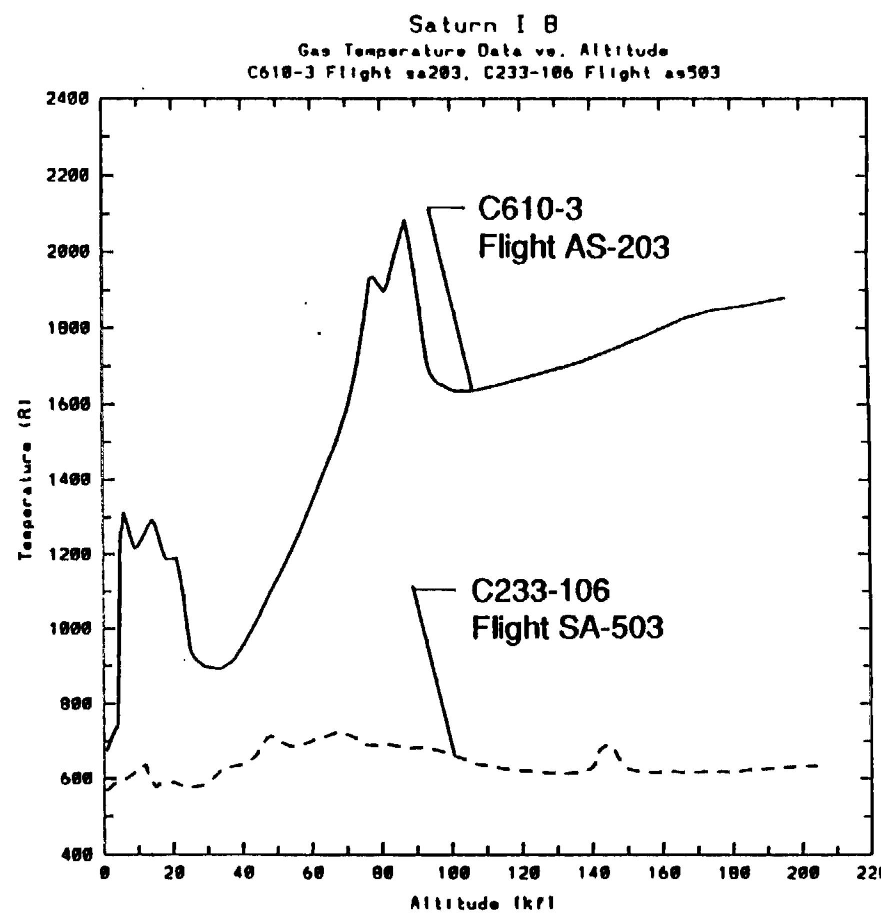
SATURN I BLOCK II TYPICAL "BAD" BASE HEATING DATA



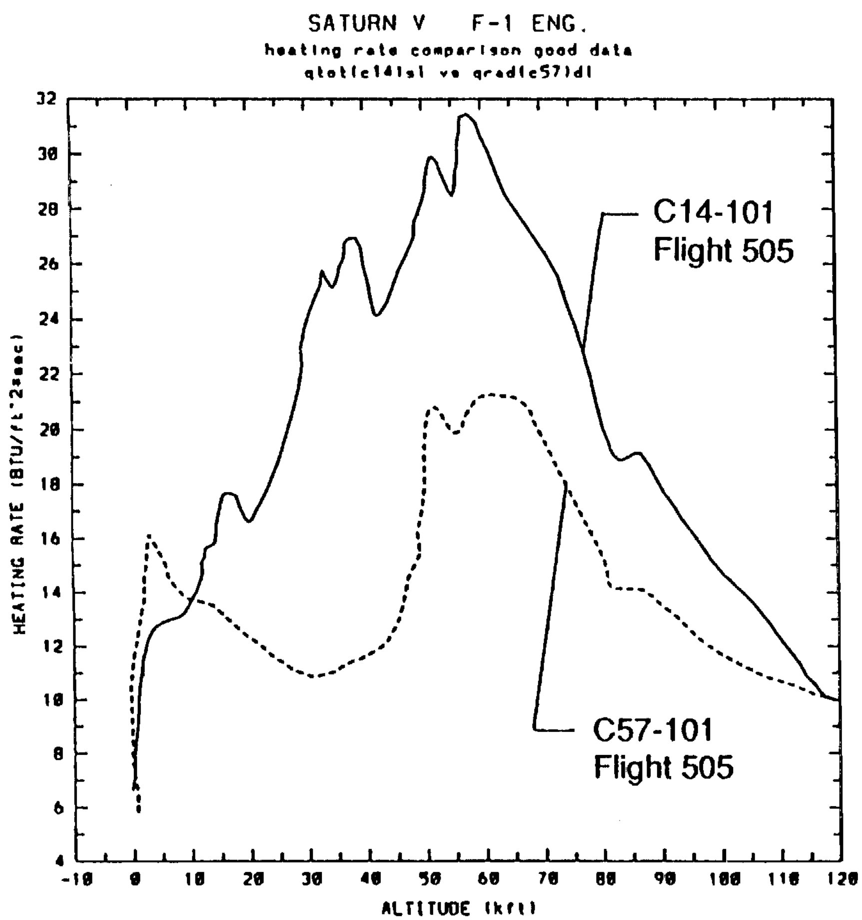


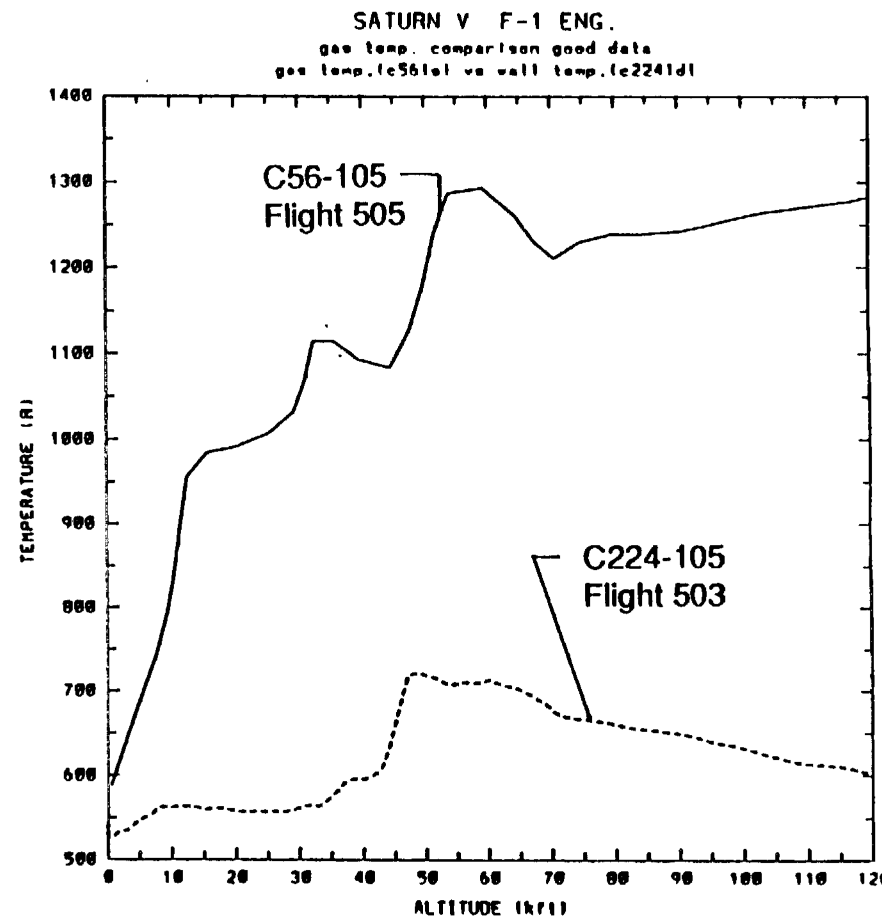
SATURN IB
TYPICAL BASE HEATING DATA



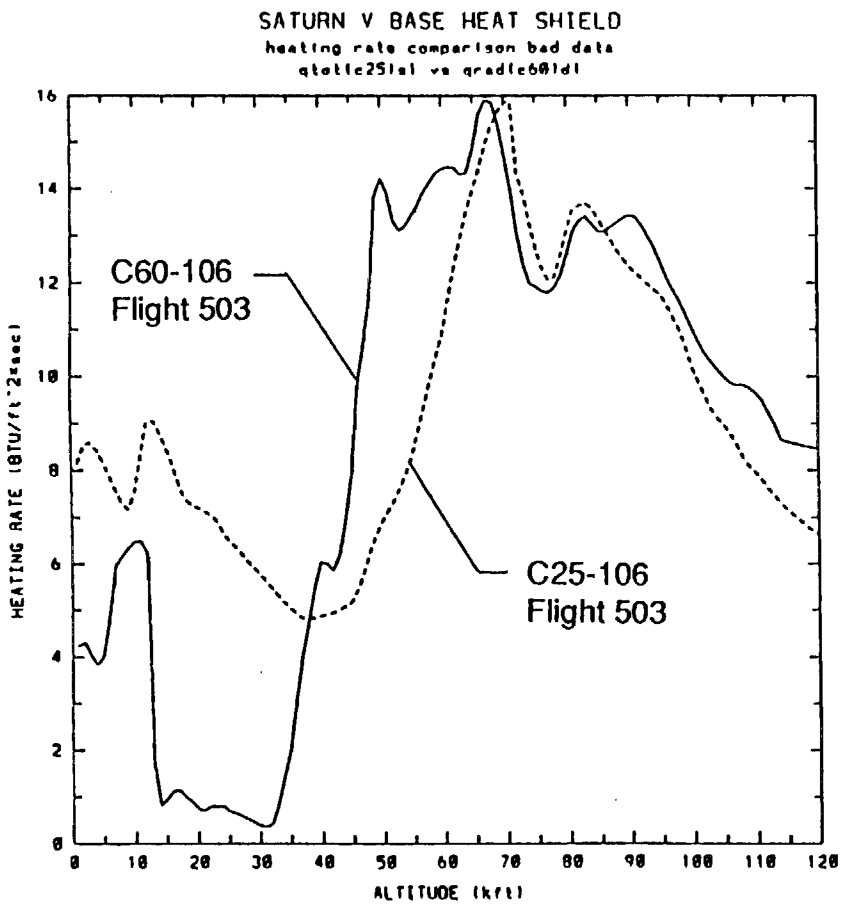


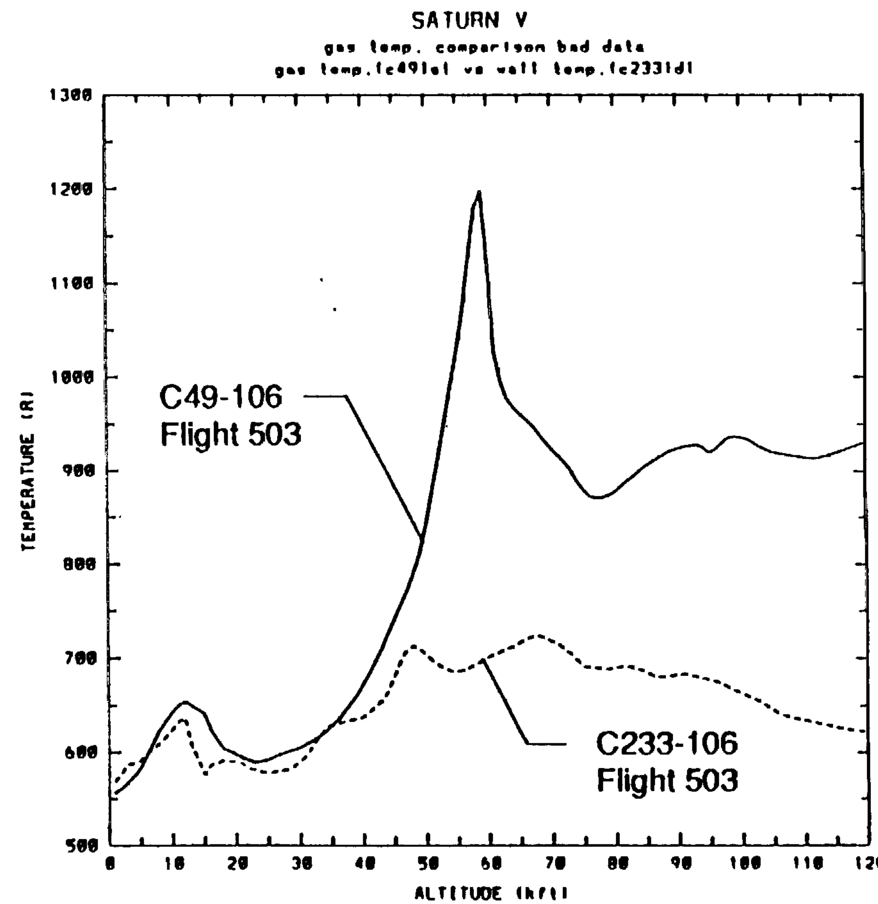
SATURN V TYPICAL "GOOD" BASE HEATING DATA



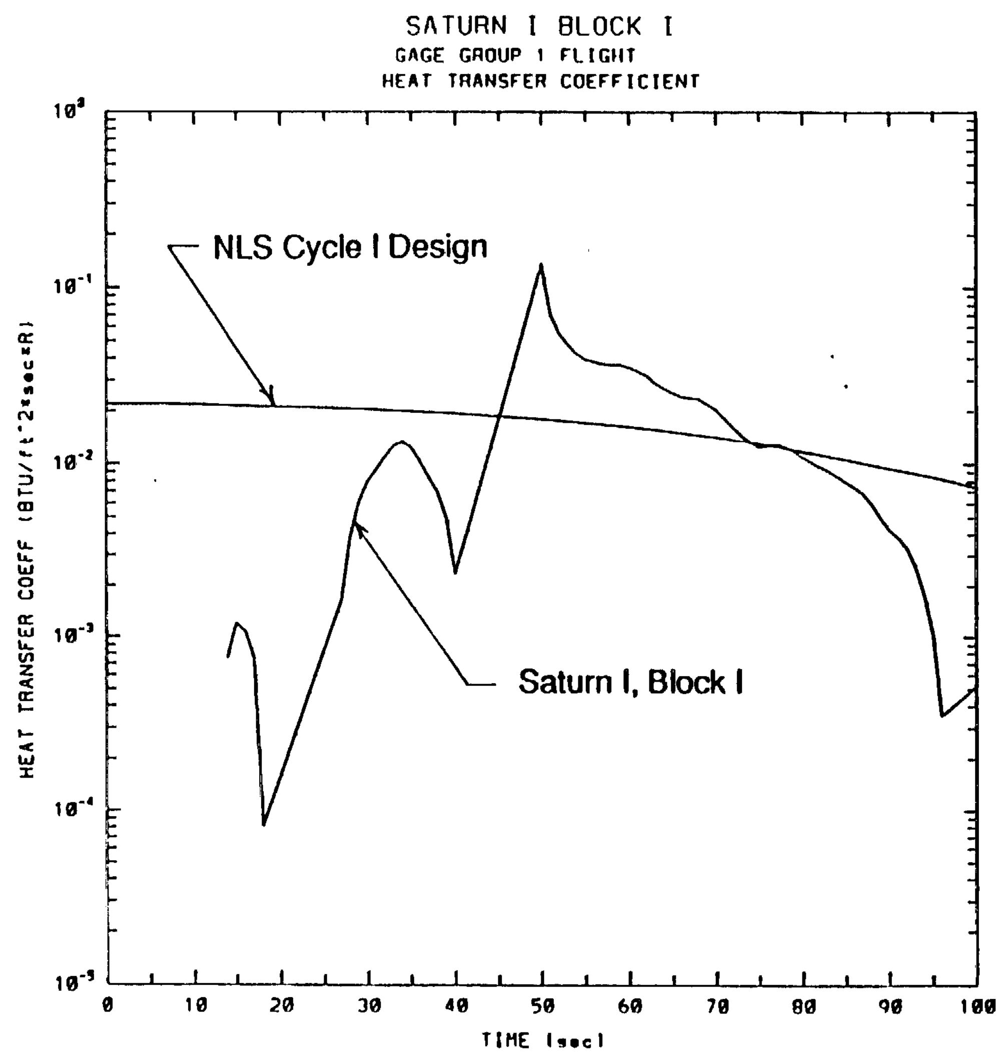


SATURN V TYPICAL "BAD" BASE HEATING DATA

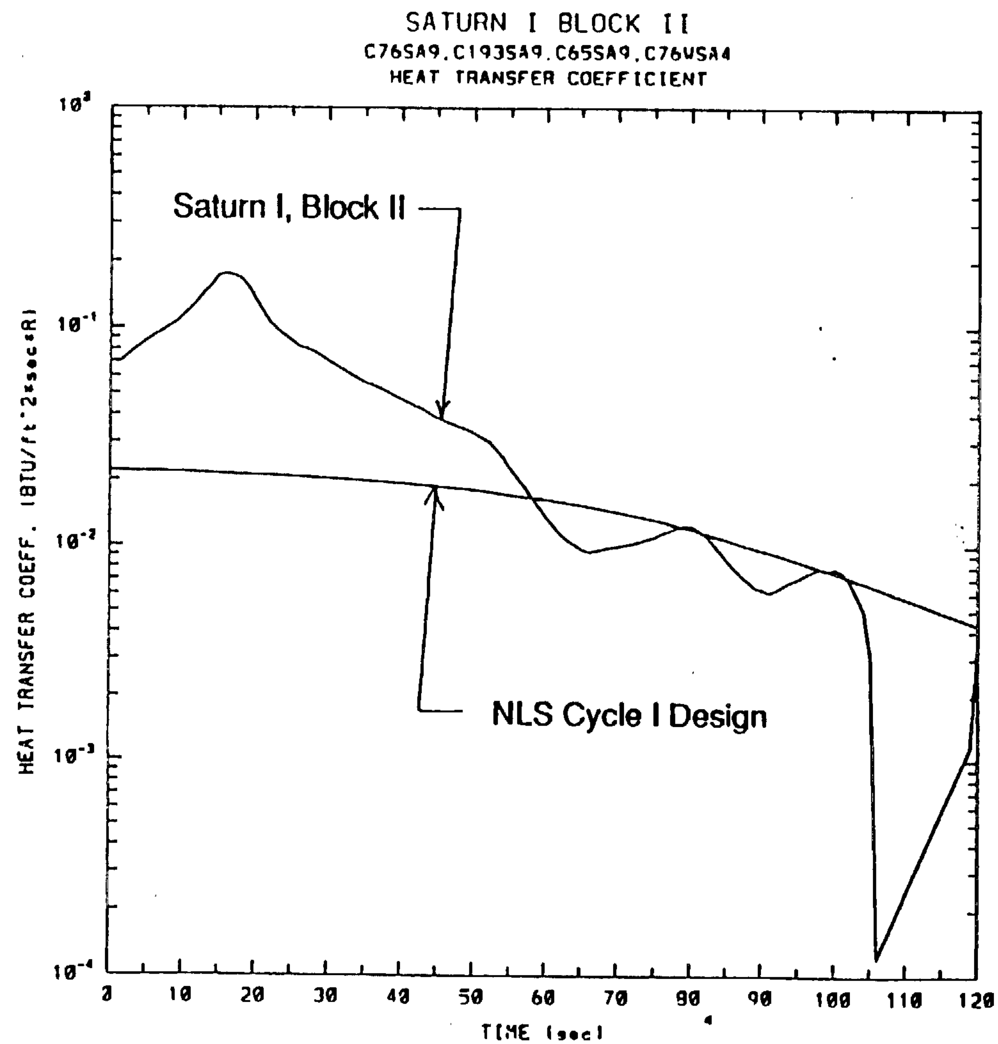




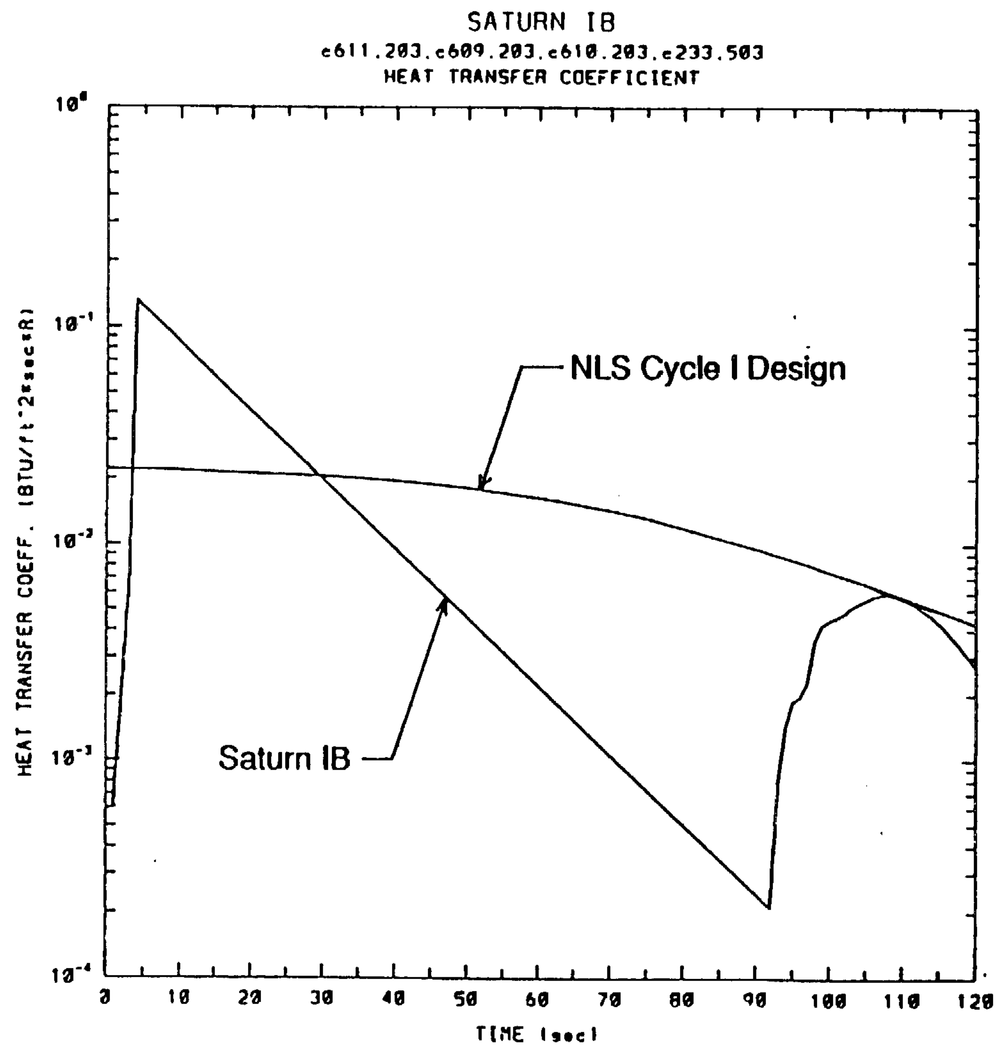
TYPICAL SATURN I BLOCK I FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



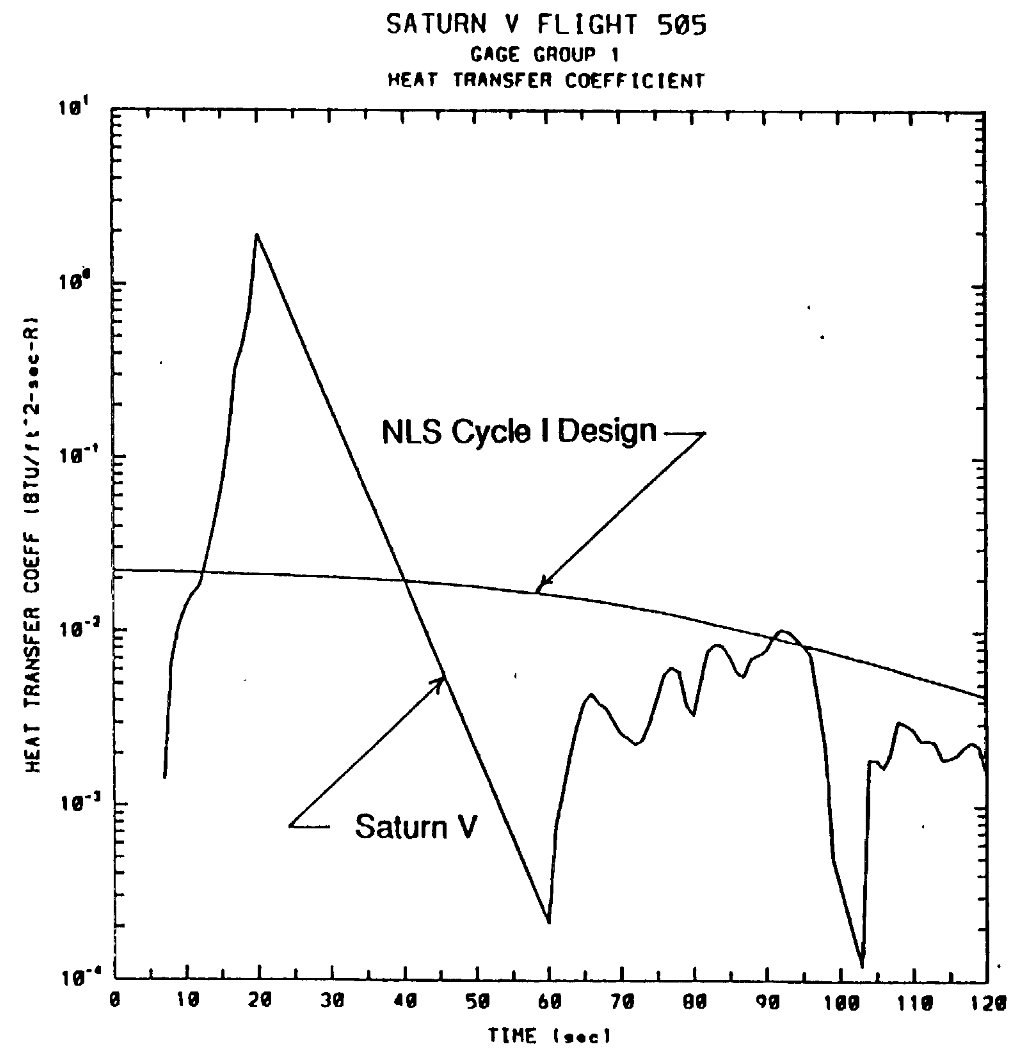
TYPICAL SATURN I BLOCK II FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



TYPICAL SATURN IB FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



TYPICAL SATURN V FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



RESULTS OF FLIGHT DEDUCED h_C

REMTECH

- Deduced h_c was meaningless in most cases because available database did not contain T_{Wall} for the total calorimeters.

- Because of lack of accurate T_{Wall}, REMTECH was unable to replicate the Cycle 1 hc design curve which was derived from Saturn I Block II data.

- The flight data does verify the general values of h_c at higher altitudes when full recirculation has occured because T_{Gas} is substantially higher than T_{Wall} (even if T_{Wall} is not precise).

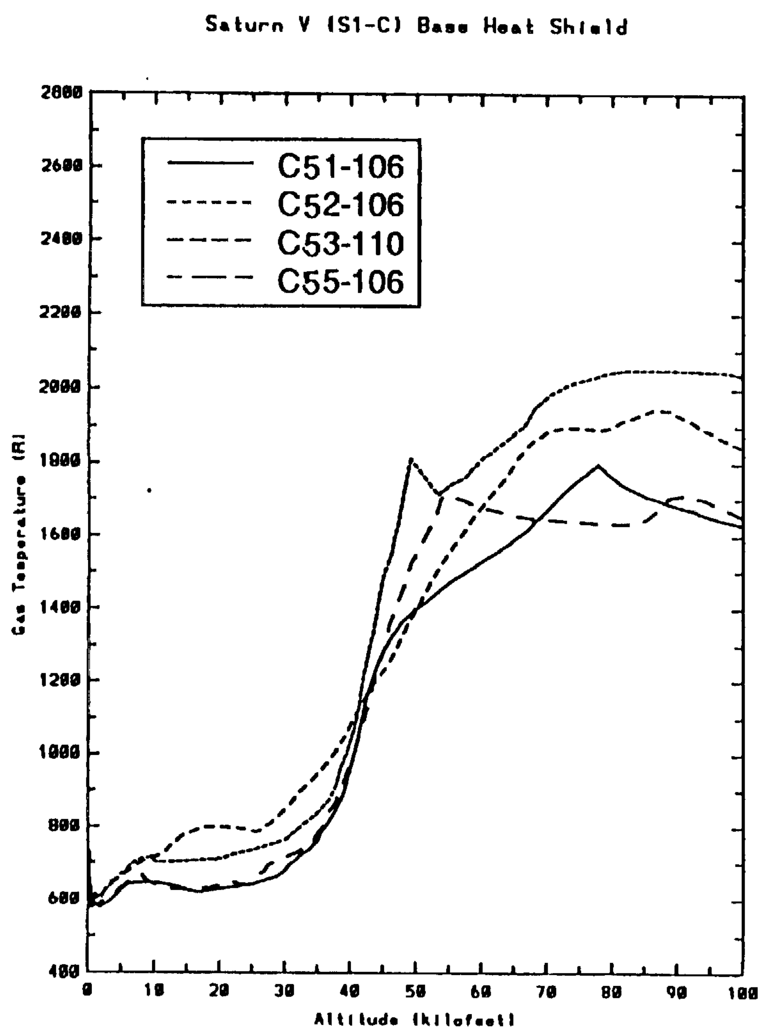
- Based on Saturn V trajectory trends, h_c is assumed valid above 35,000 feet or about 75 seconds into flight.

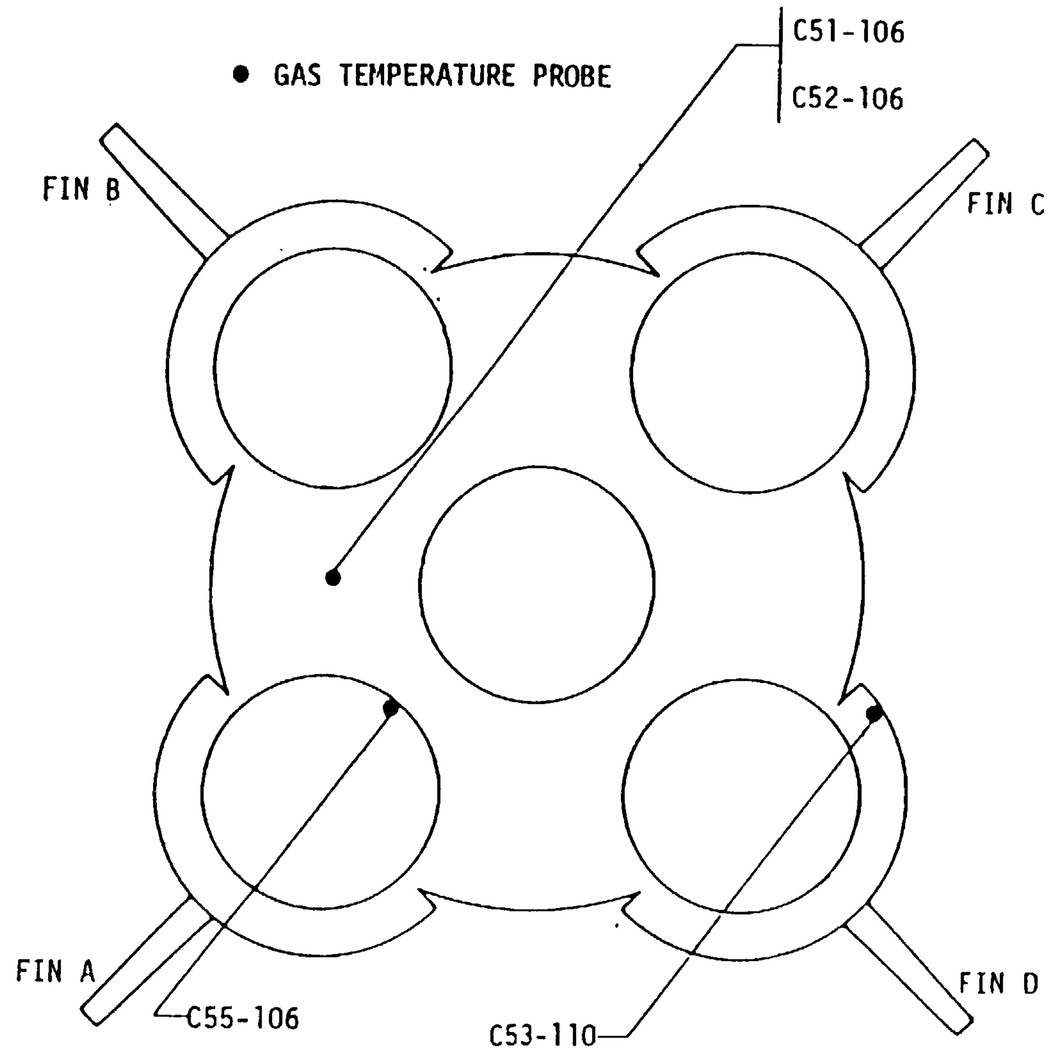
- A technique to adjust h_c to lower altitudes is provided in application section of this presentation.

TASK 2 - T_{GAS} FROM SATURN FLIGHT DATA TRENDS

SATURN V GAS TEMPERATURE VARIATION WITH BASE HEAT SHIELD LOCATION

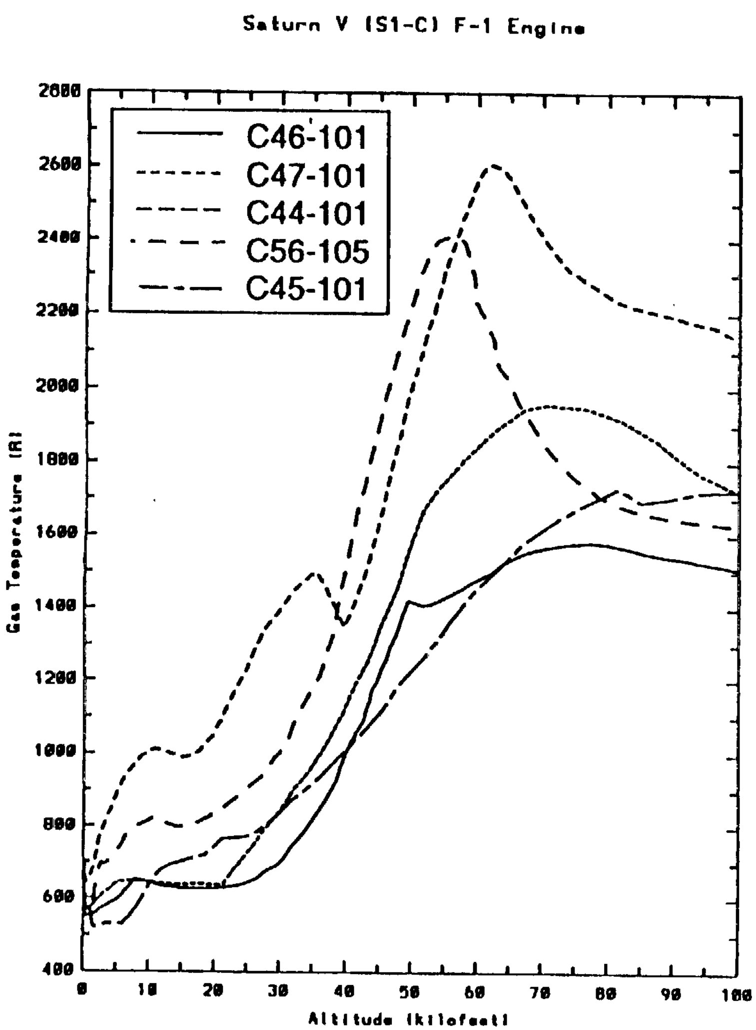
FLIGHT AS-502

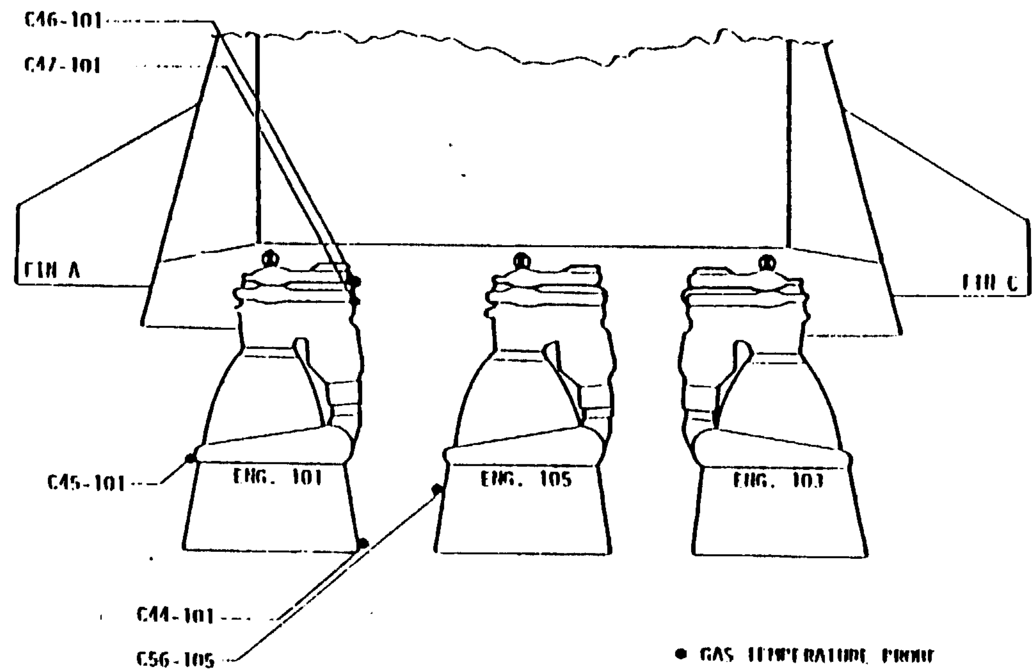




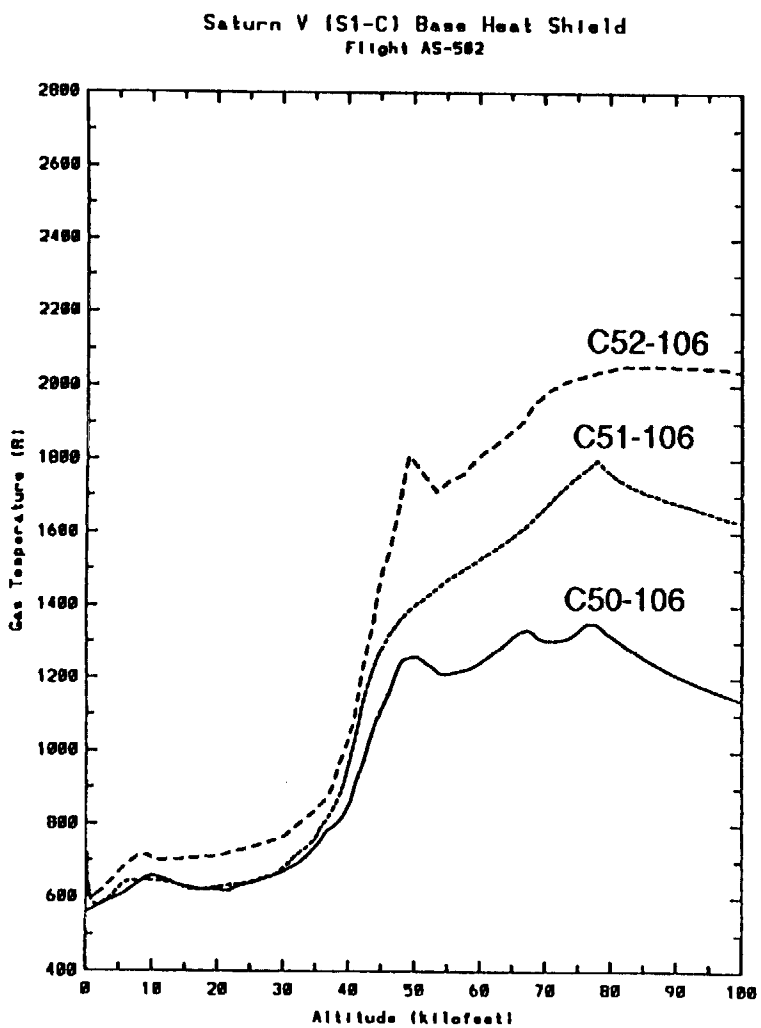
SATURN V GAS TEMPERATURE VARIATION WITH F-1 ENGINE LOCATION

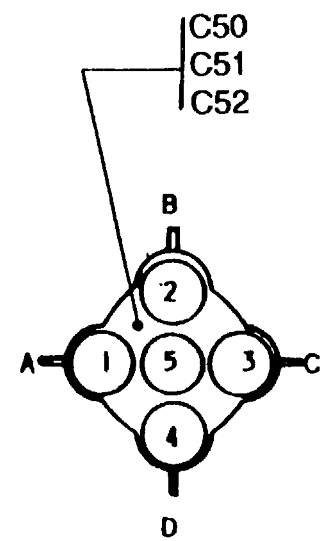
FLIGHT AS-502





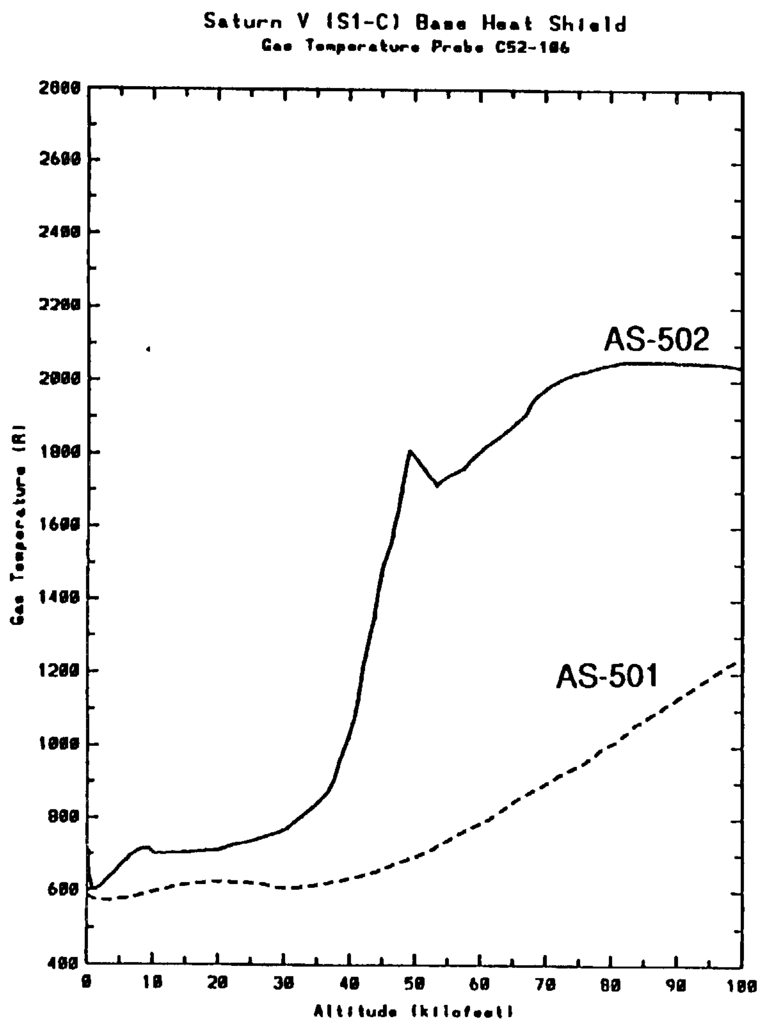
SATURN V GAS TEMPERATURE EFFECT OF PROBE HEIGHT OFF HEAT SHIELD SURFACE

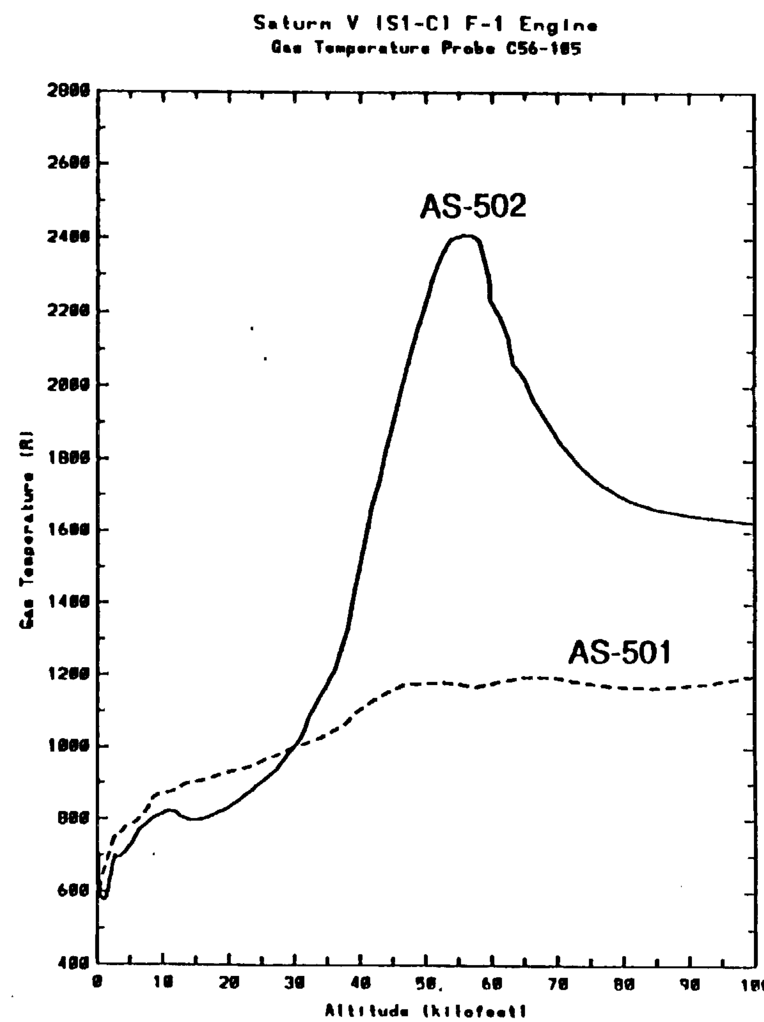




Probe	Height Off Surface
C50	0.25"
C51	1.00"
C52	2.50"

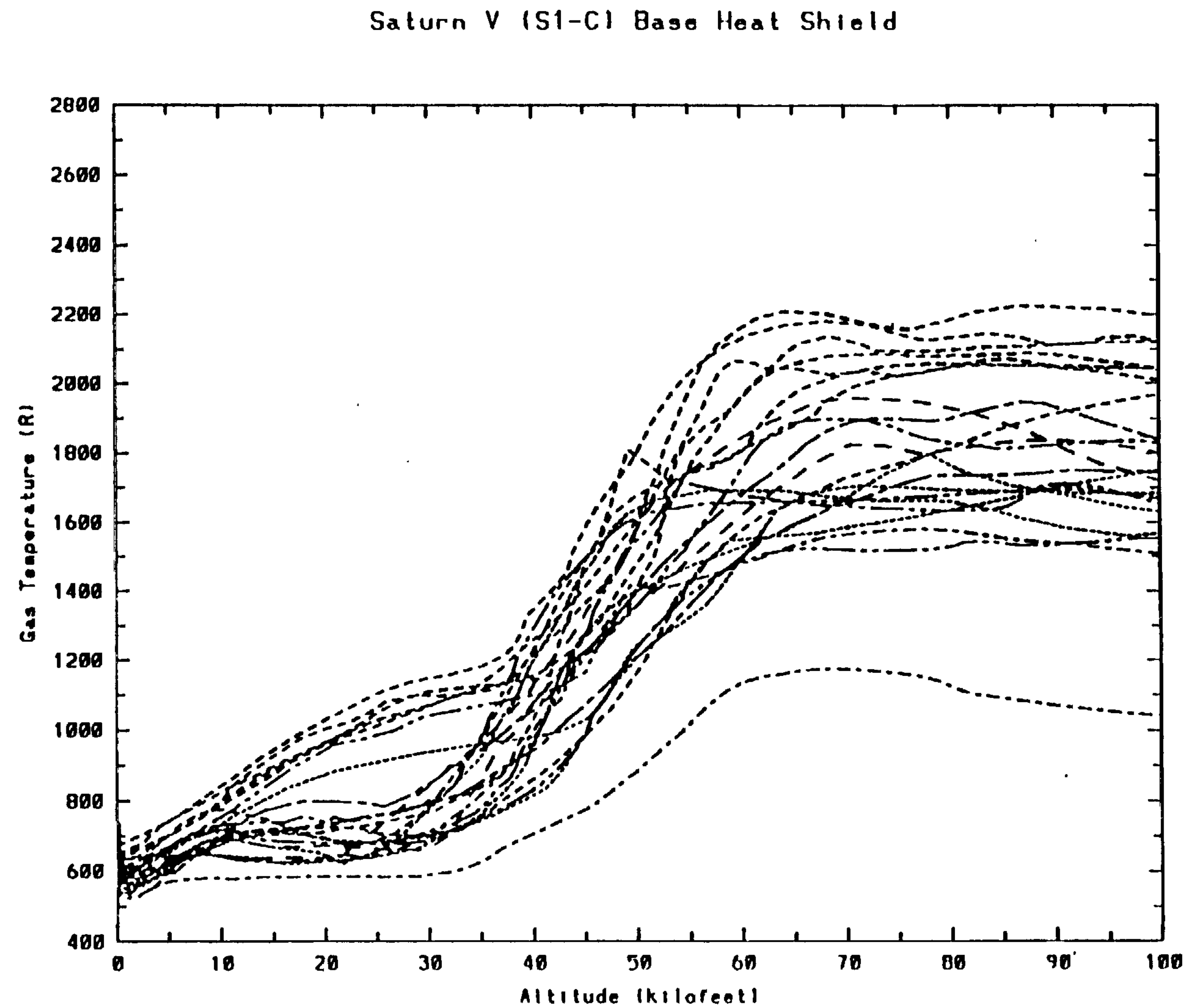
SATURN V GAS TEMPERATURE FLOW DEFLECTOR EFFECT
AS-501 vs AS-502





SATURN V GAS TEMPERATURE BASE HEAT SHIELD DATA FLIGHTS AS-502 - AS-509

22 FLIGHT MEASUREMENTS

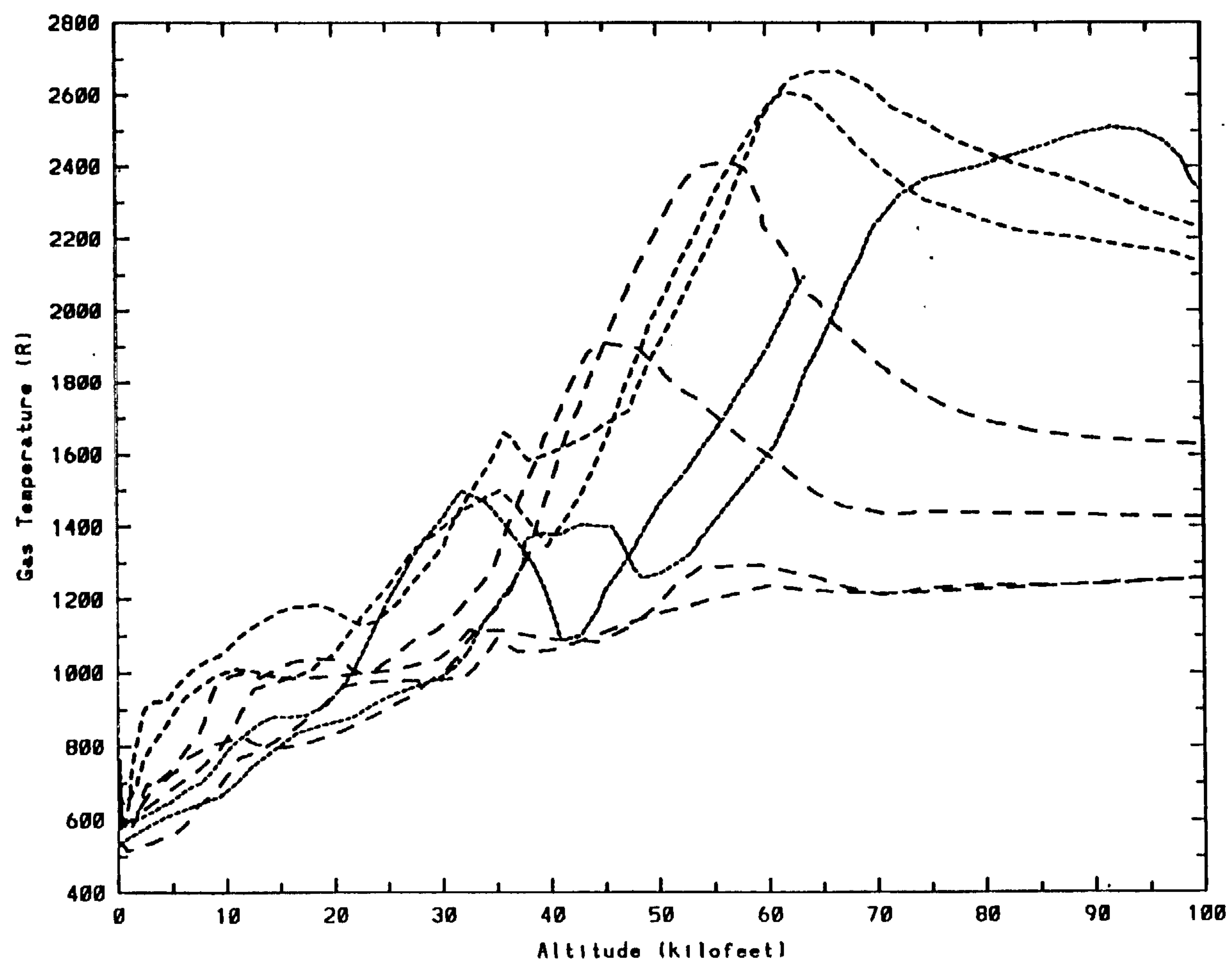


SATURN V GAS TEMPERATURE F-1 ENGINE
DATA FLIGHTS AS-502 - AS-505

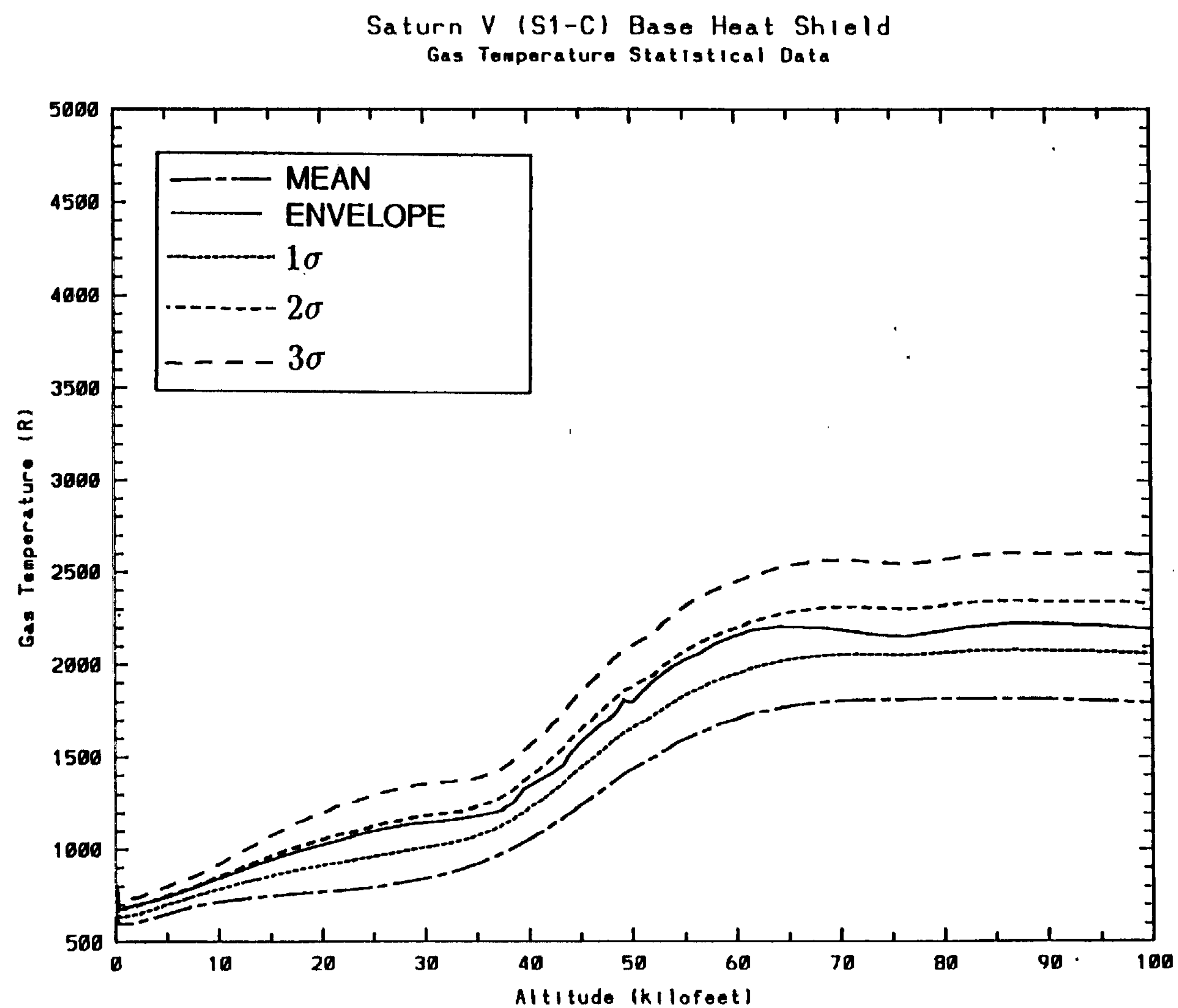
REMTECH

8 FLIGHT MEASUREMENTS

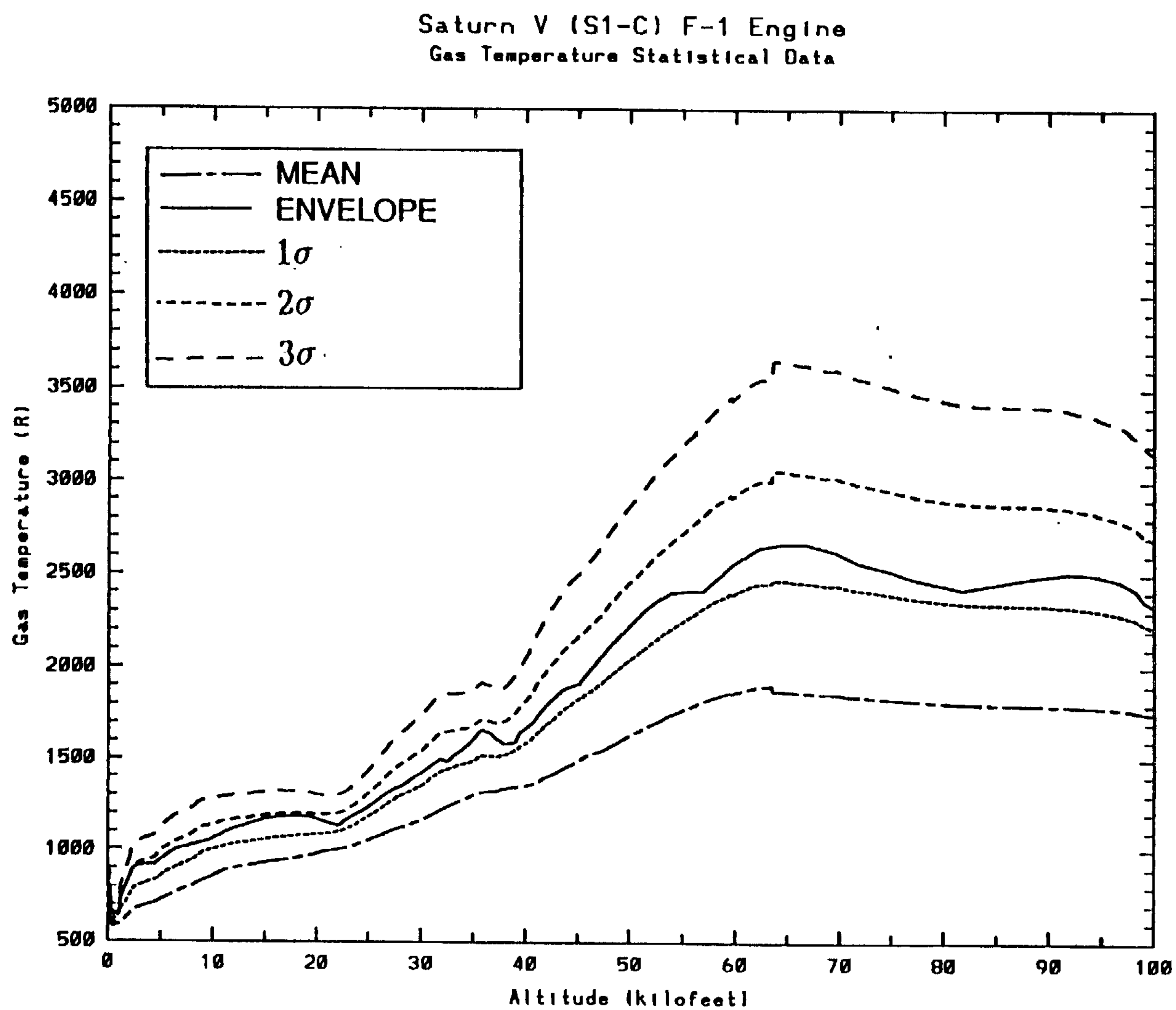
Saturn V (S1-C) F-1 Engine



SATURN V GAS TEMPERATURE BASE HEAT SHIELD STATISTICAL DATA

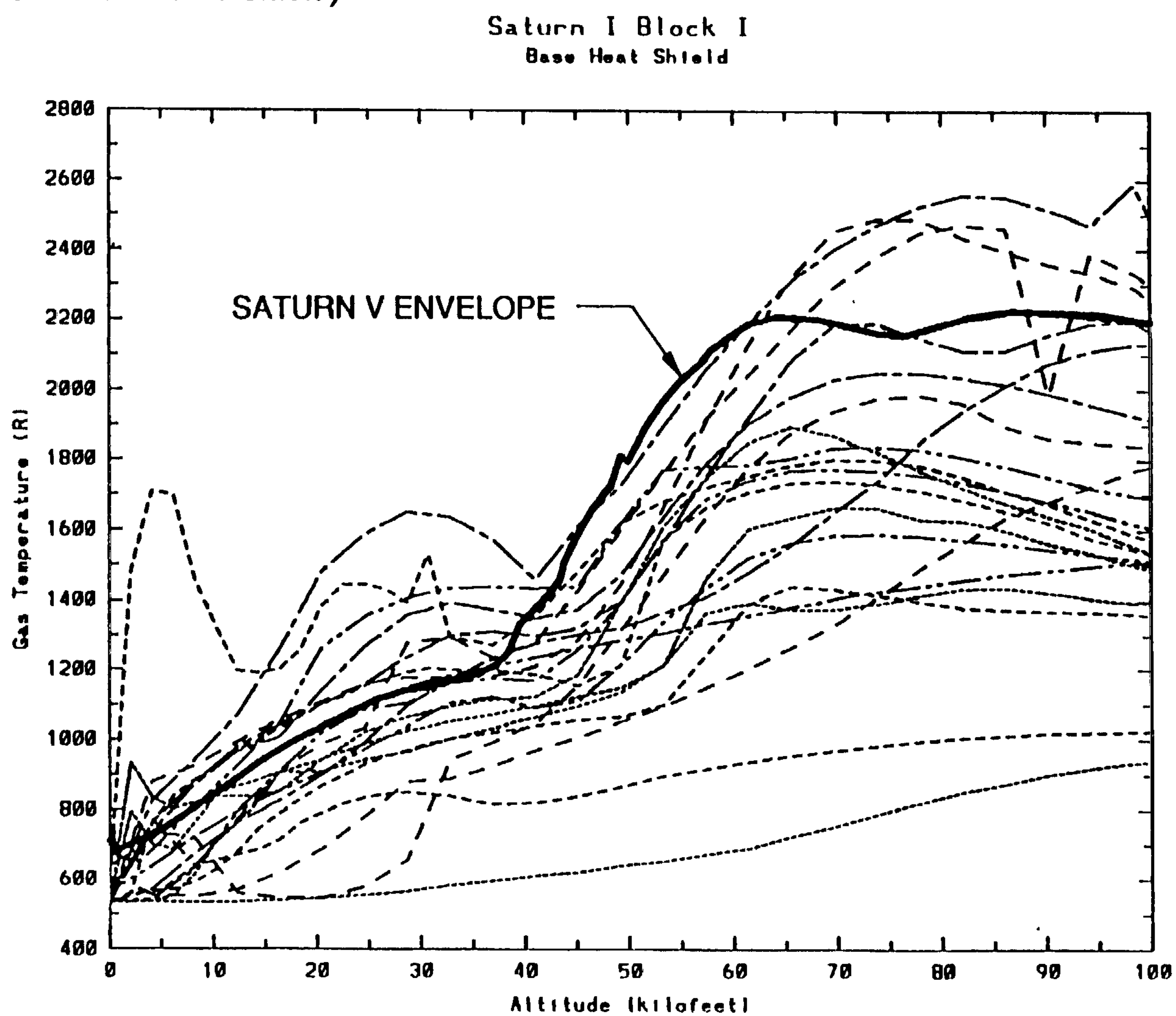


SATURN V GAS TEMPERATURE
F-1 ENGINE STATISTICAL DATA



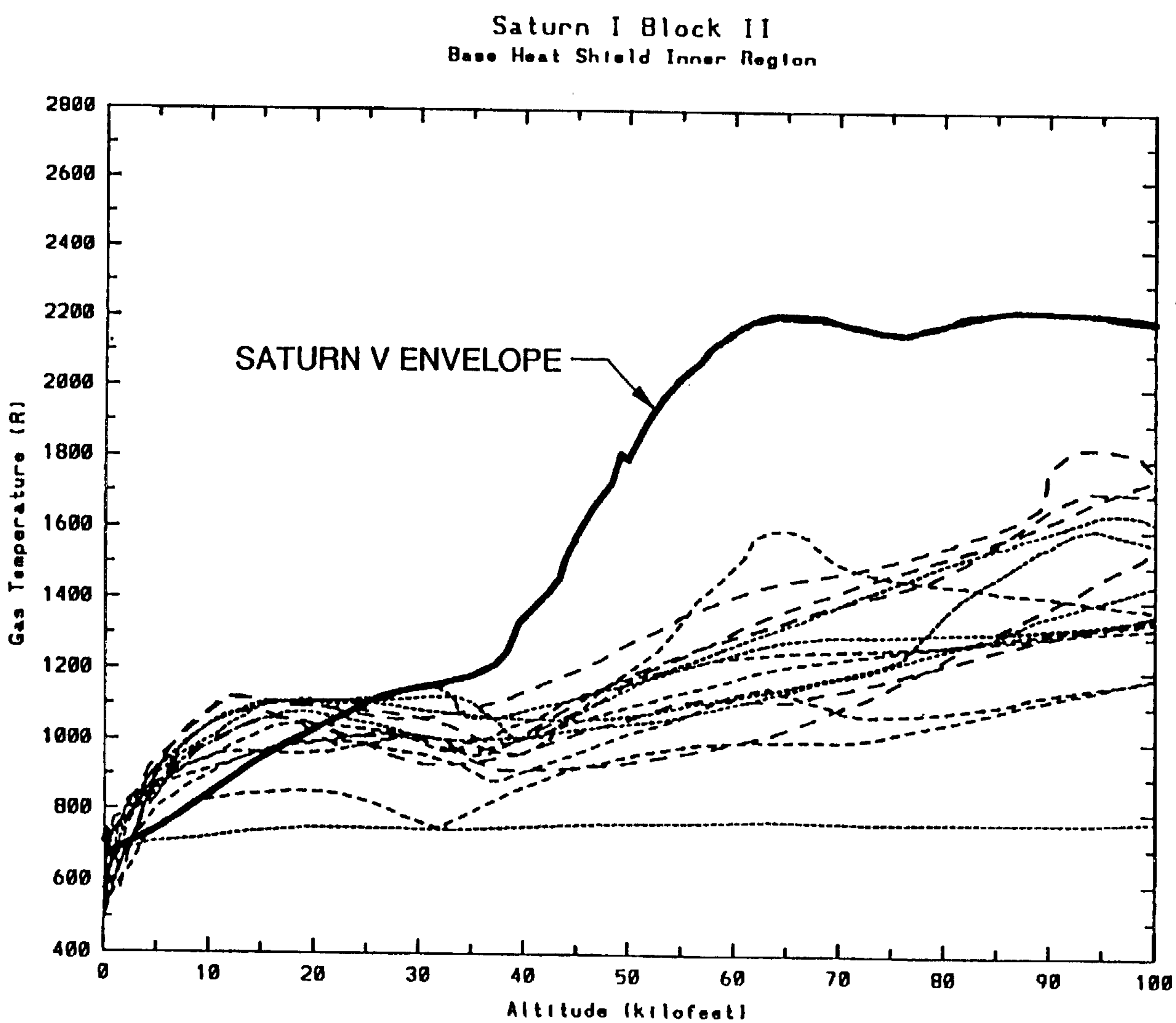
SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK I

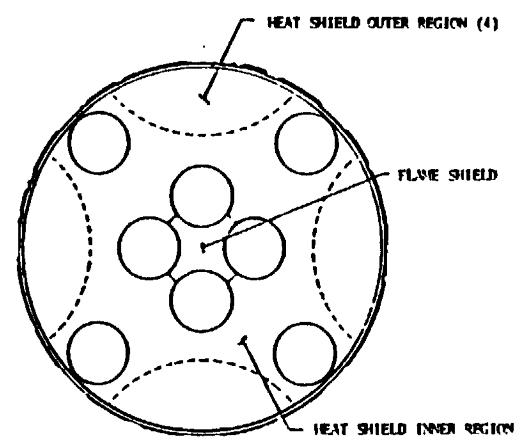
19 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK I
(Does not include Flame Shield)



SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK II INNER REGION

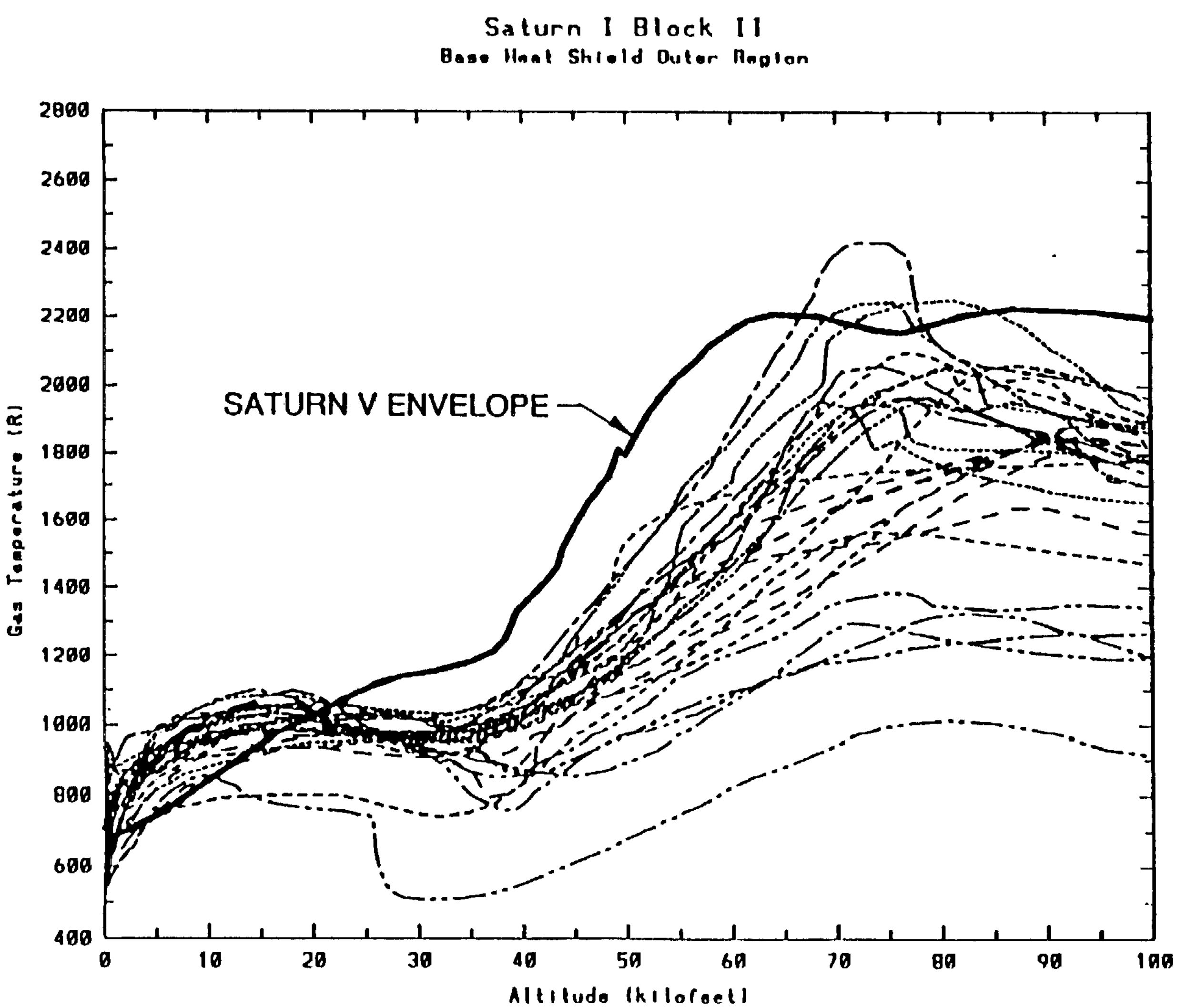
14 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK II INNER REGION

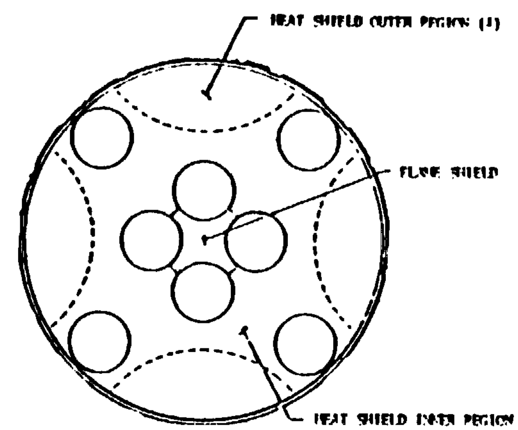




SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK II OUTER REGION

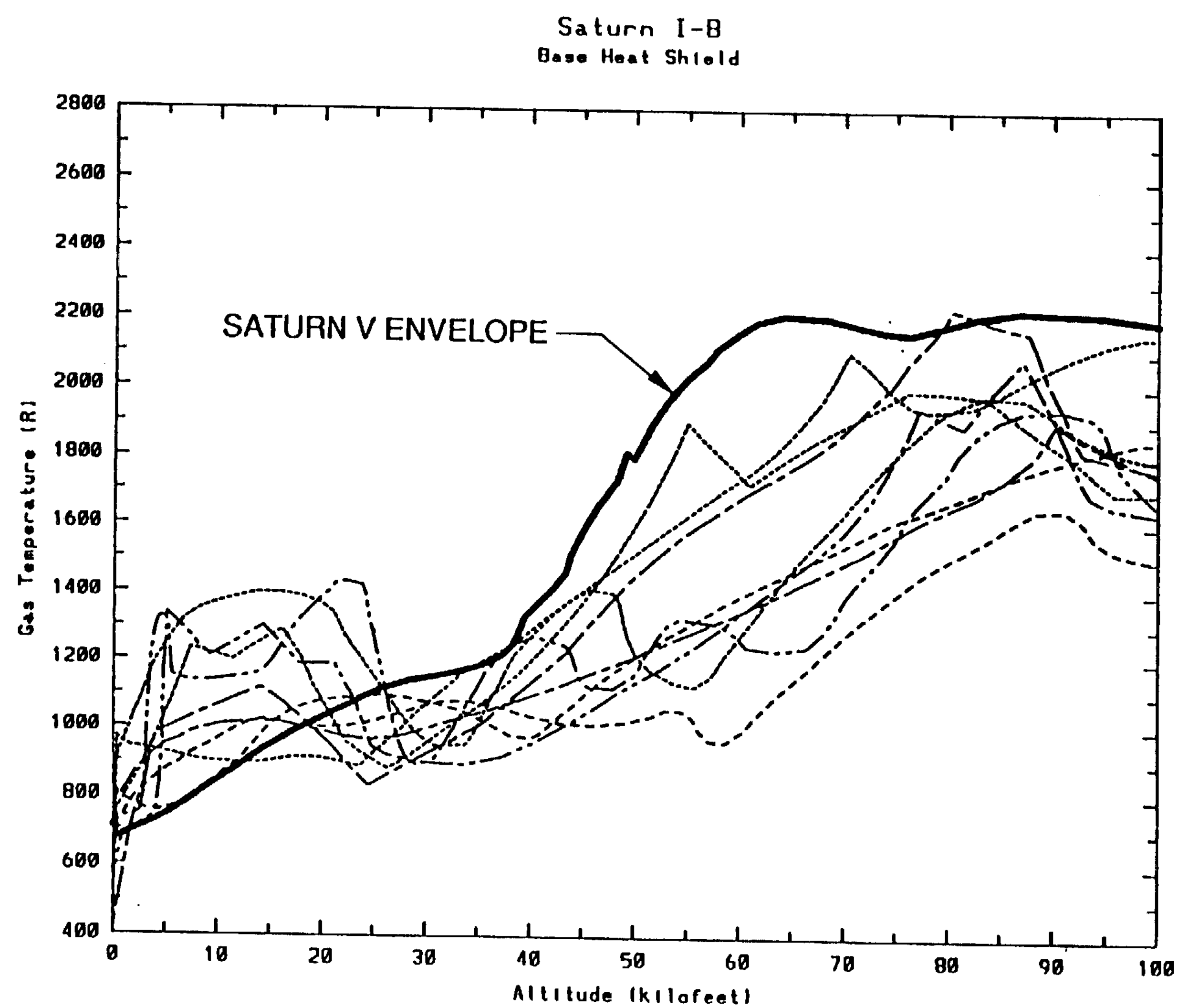
28 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK II OUTER REGION





SATURN GAS TEMPERATURE BASE HEAT SHIELD SATURN V vs SATURN IB

9 FLIGHT MEASUREMENTS FROM SATURN IB



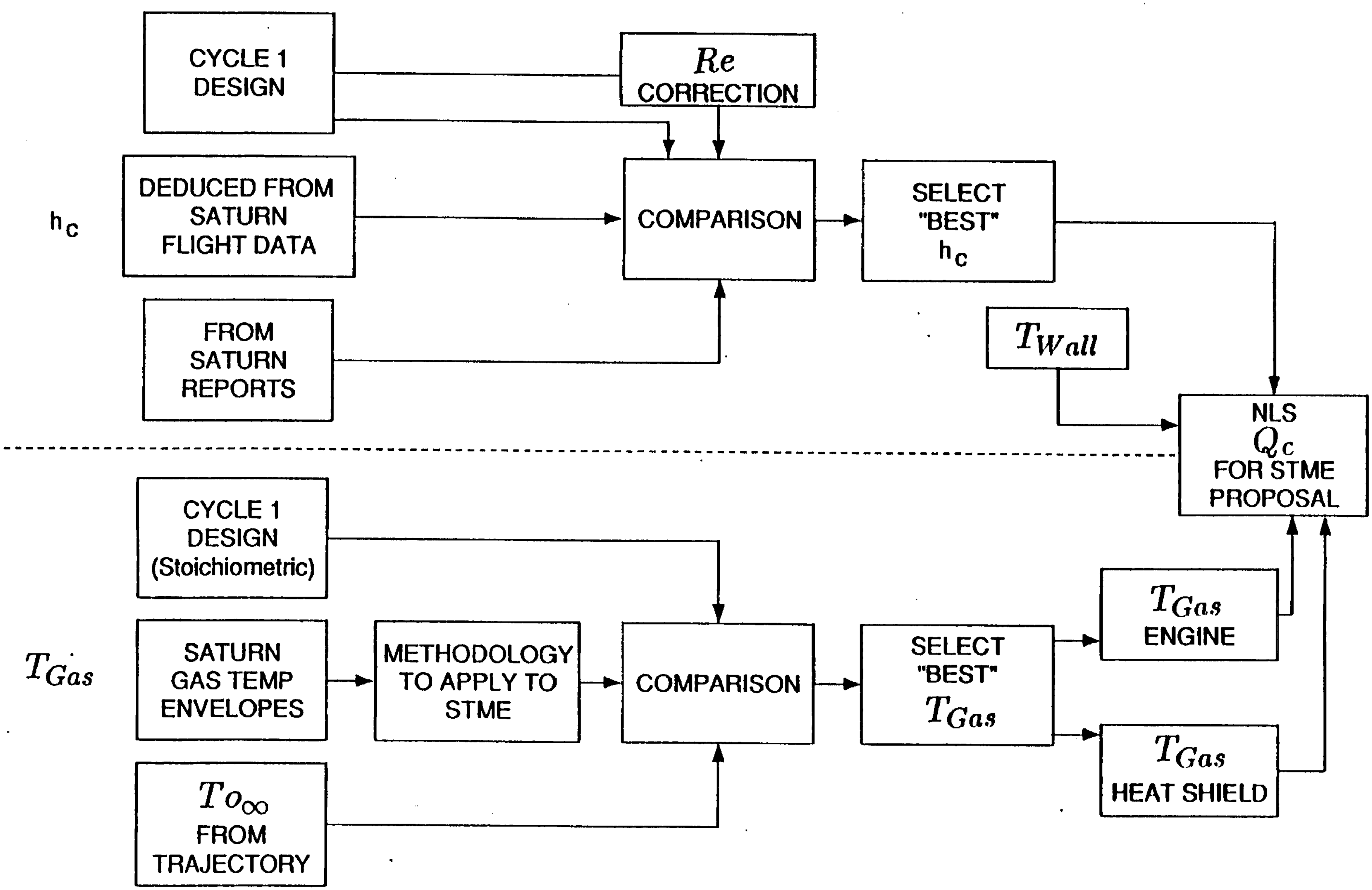
RESULTS OF SATURN V GAS TEMPERATURE REVIEW

REMTECH

- The gas temperature data are relatively consistent, repeatable, and high quality.
- Data have been separated into two main groups: base heat shield and inboard engine surfaces.
- AS-501 data are not included because of flow deflector effect.
- Envelopes of all data were determined as well as statistical mean and 1σ, 2σ, and 3σ standard deviations.
- Saturn V data does ***not*** envelope Saturn I data below 20,000 feet.
- Saturn V gas temperatures (excluding AS-501) are greater than freestream total temperature up to 100,000 feet.
- A methodology to deduce air/turbine exhaust mixture ratios from these data is presented in the applications section.

APPLICATION OF RESULTS TO NLS 1.5 STAGE VEHICLE

APPLICATION OF SATURN REVIEW RESULTS TO NLS 1.5 STAGE VEHICLE



METHODOLOGY FOR IMPROVING h_c EARLY IN FLIGHT

REMTECH

BASE REGION REYNOLDS NUMBER ADJUSTMENT

Steps:

1. Assume h_c Cycle 1 Design curve valid for Saturn V S-1C base region after plume-to-plume recirculation begins at 12 Km altitude

2. For typical Saturn V trajectory, 12 Km is reached about 75 seconds into flight at a vehicle Mach number of approximately 1.6

3. Assume $M_B \approx 0.2 M_\infty$ and compute $(\rho_B V_B)^{0.8}$ at t = 75 seconds

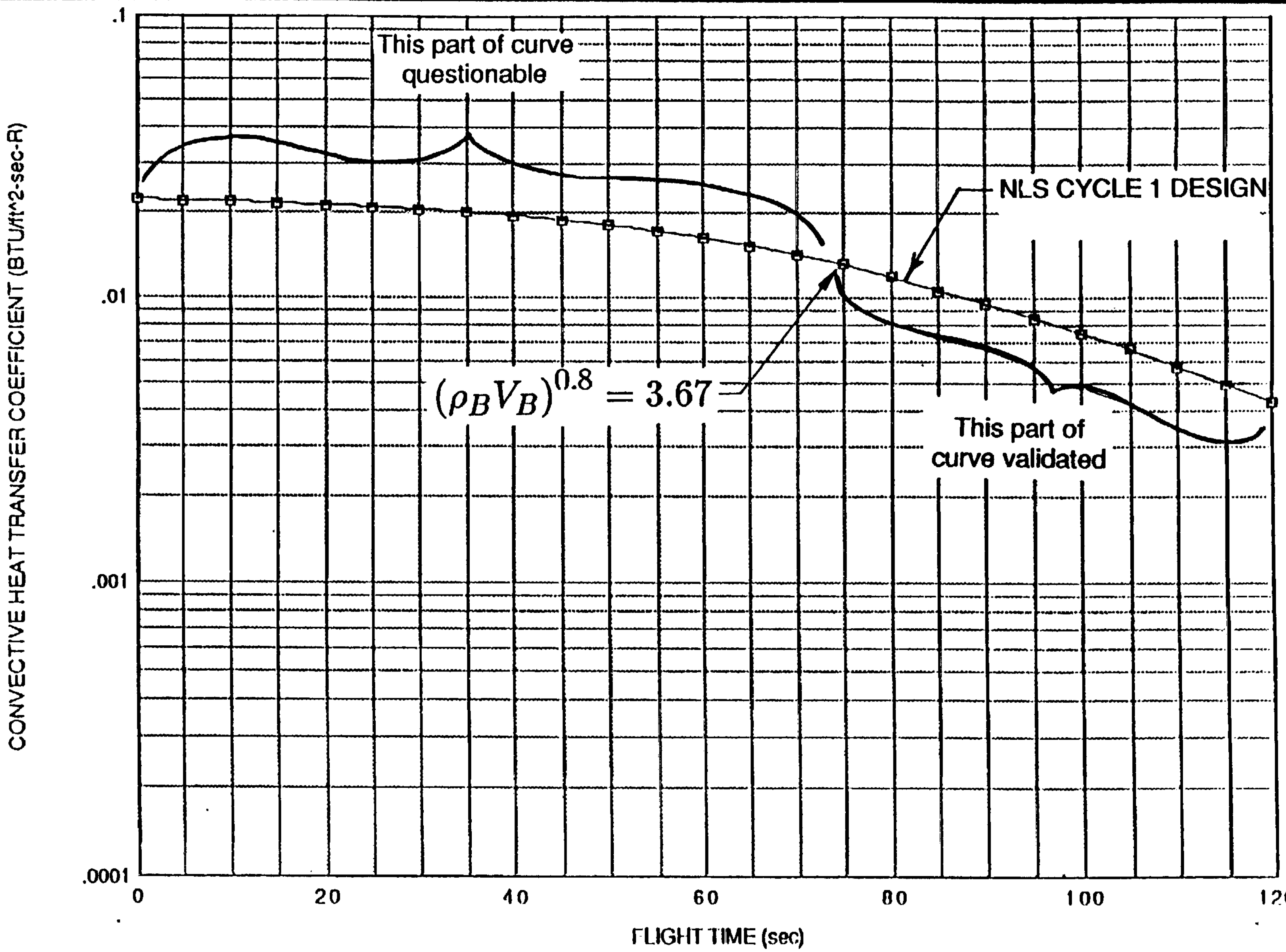
4. Compute $(\rho_B V_B)^{0.8}$ for time = 0 to 75 seconds from Saturn V trajectory assuming $P_B = P_\infty$, $T_B = To_\infty$, and $M_B = 0.2 M_\infty$

5. Compute ratio: $\dfrac{(\rho_B V_B)^{0.8}}{(\rho_B V_B)^{0.8}_{t=75}}$

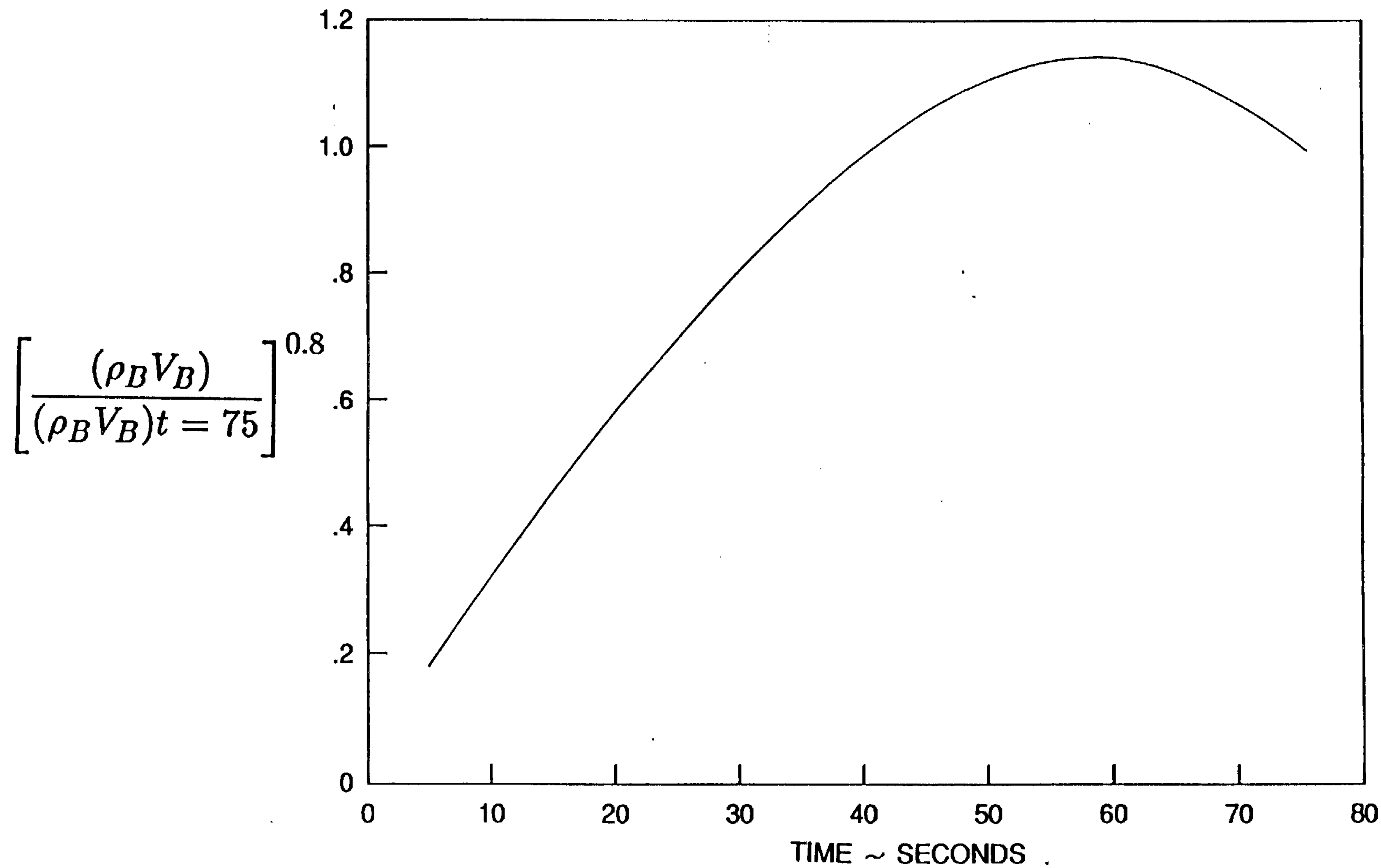
6. Apply ratio to h_c @ t = 75 to define h_c from t = 0 to t = 75

7. Vary M_B assumption and $\rho_B V_B$ exponent to determine sensitivity

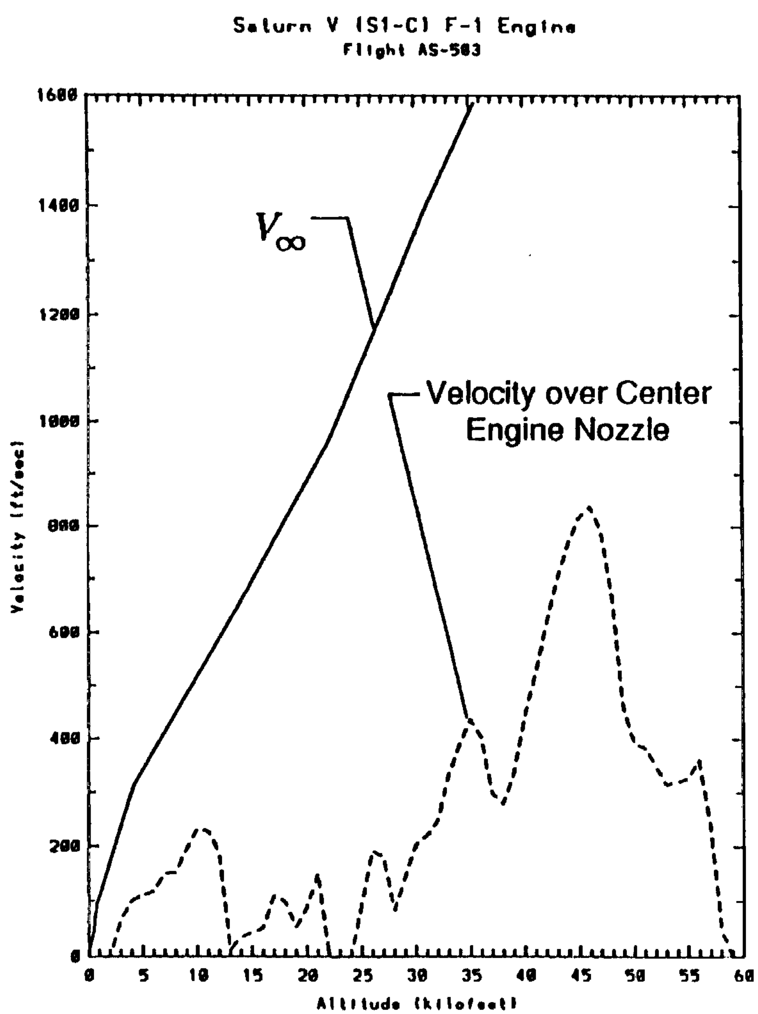
REYNOLDS NUMBER ADJUSTMENT FOR h_c

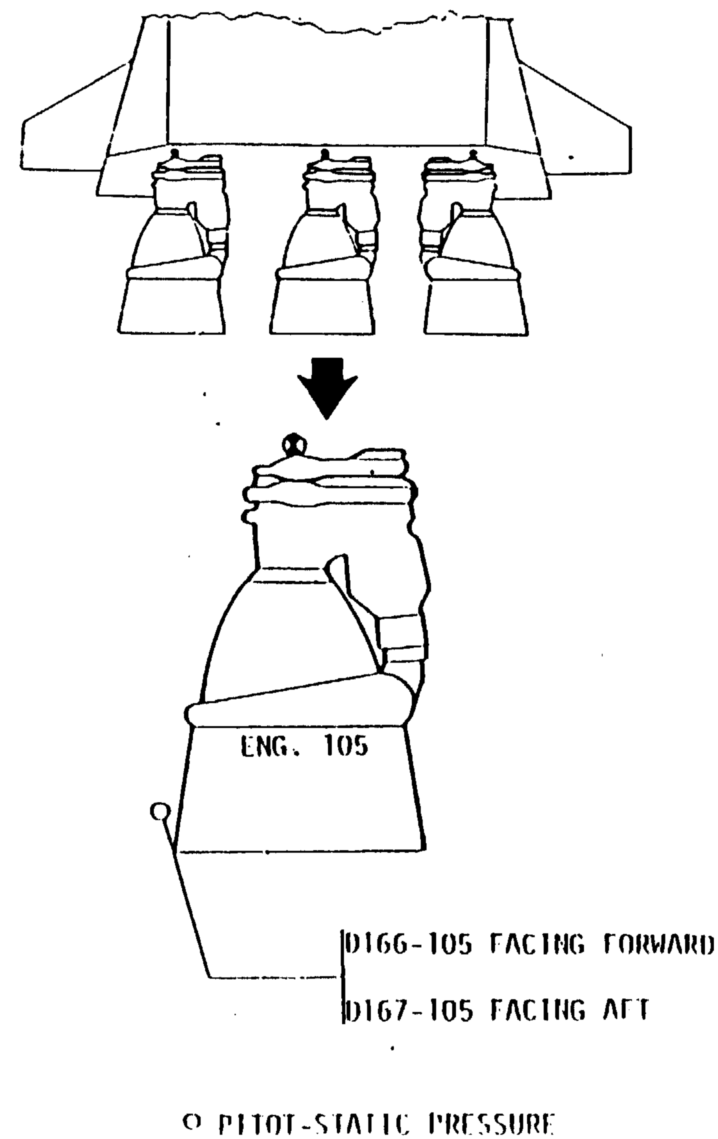


TRAJECTORY EFFECT ON BASE REGION REYNOLDS NUMBER

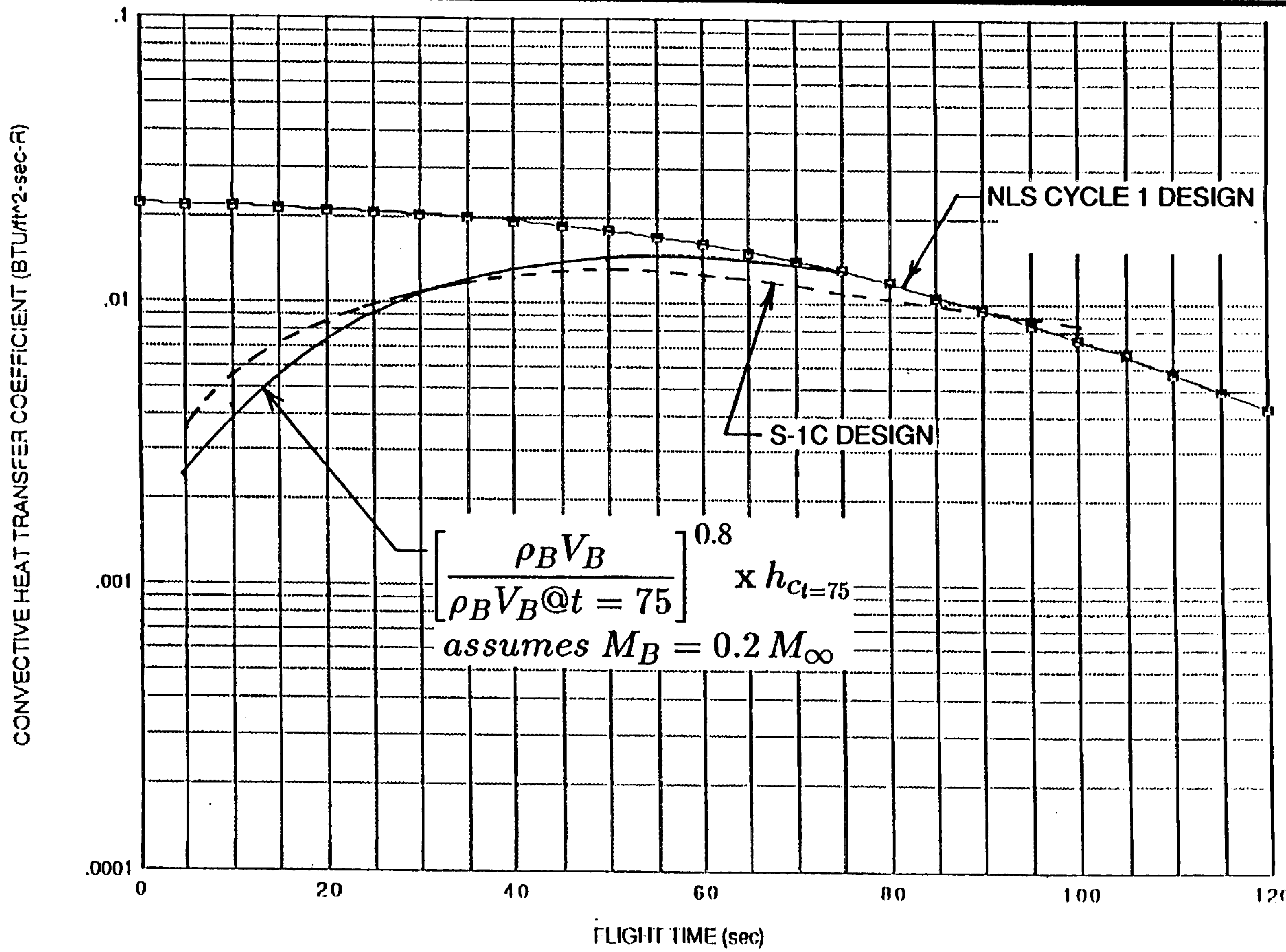


SATURN V BASE REGION VELOCITY
DEDUCED FROM CENTER ENGINE PRESSURE DATA

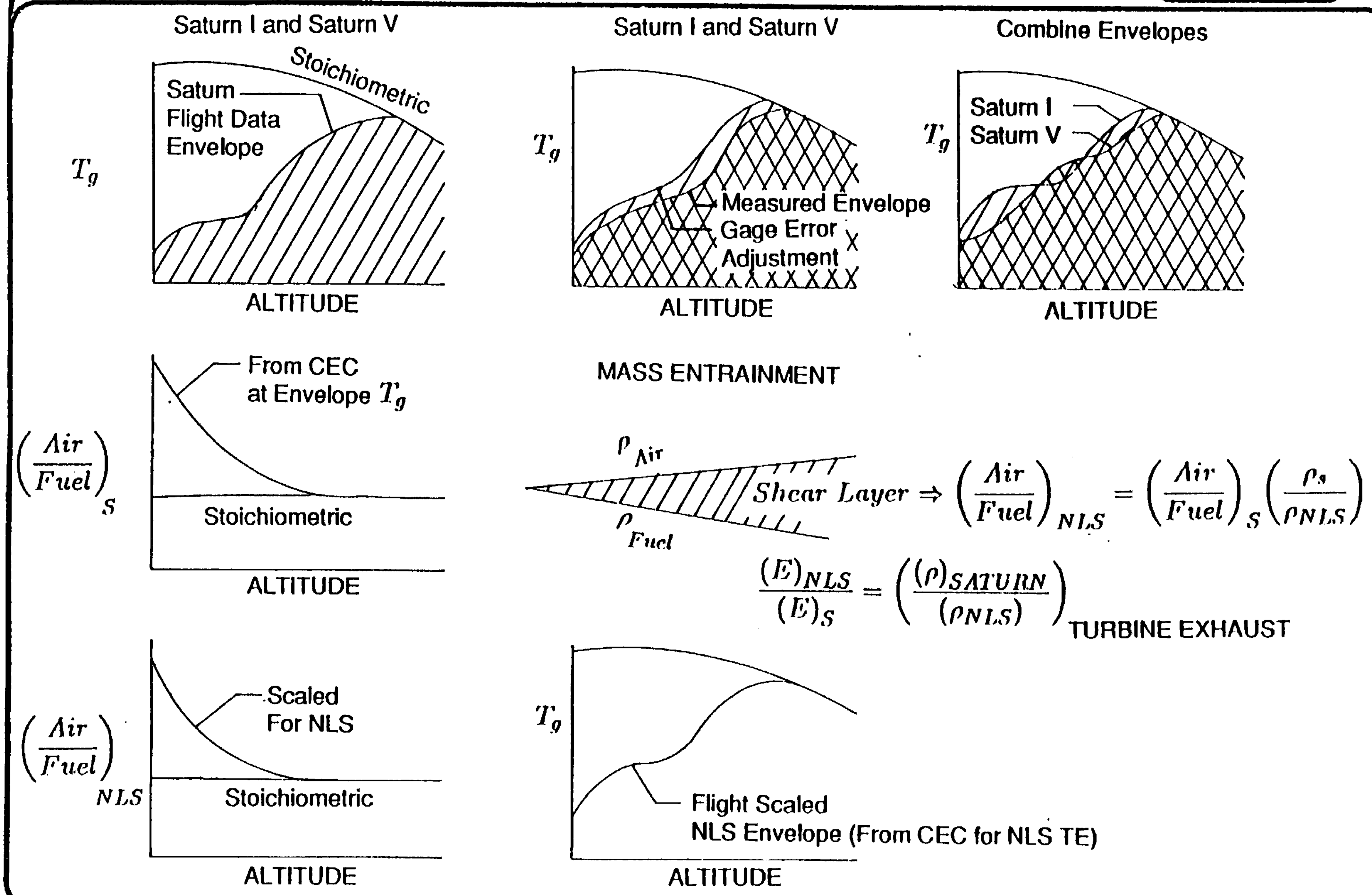




BASE REYNOLDS NUMBER ADJUSTMENT TO CYCLE 1 DESIGN h_c



METHODOLOGY FOR APPLYING SATURN DATA TO NLS GAS TEMPERATURE



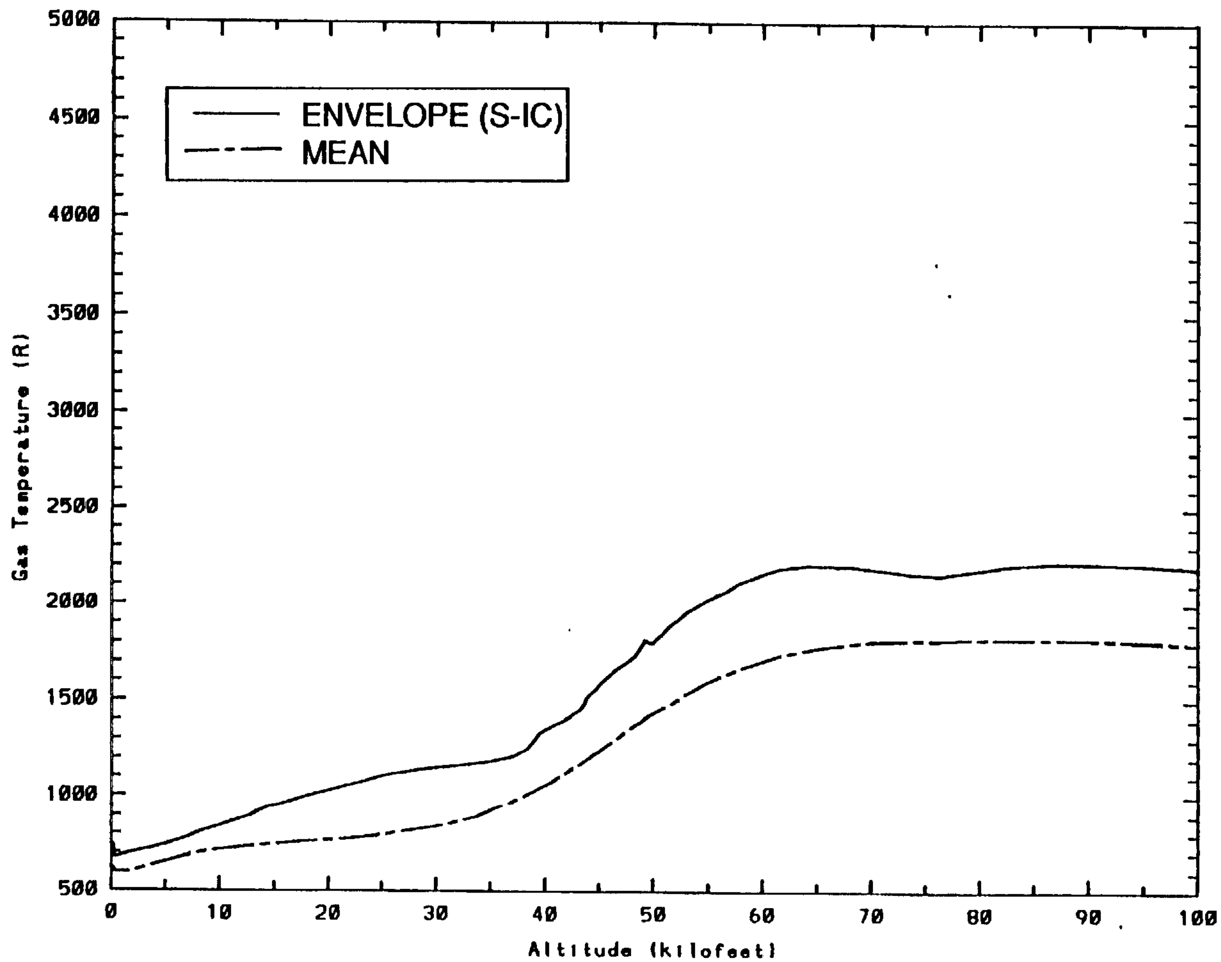
$$Shear\ Layer \Rightarrow \left(\frac{Air}{Fuel}\right)_{NLS} = \left(\frac{Air}{Fuel}\right)_S \left(\frac{\rho_s}{\rho_{NLS}}\right)$$

$$\frac{(E)_{NLS}}{(E)_S} = \left(\frac{(\rho)_{SATURN}}{(\rho_{NLS})}\right)_{\text{TURBINE EXHAUST}}$$

SATURN V S-1C STAGE BASE HEAT SHIELD FLIGHT TEMPERATURE HISTORIES

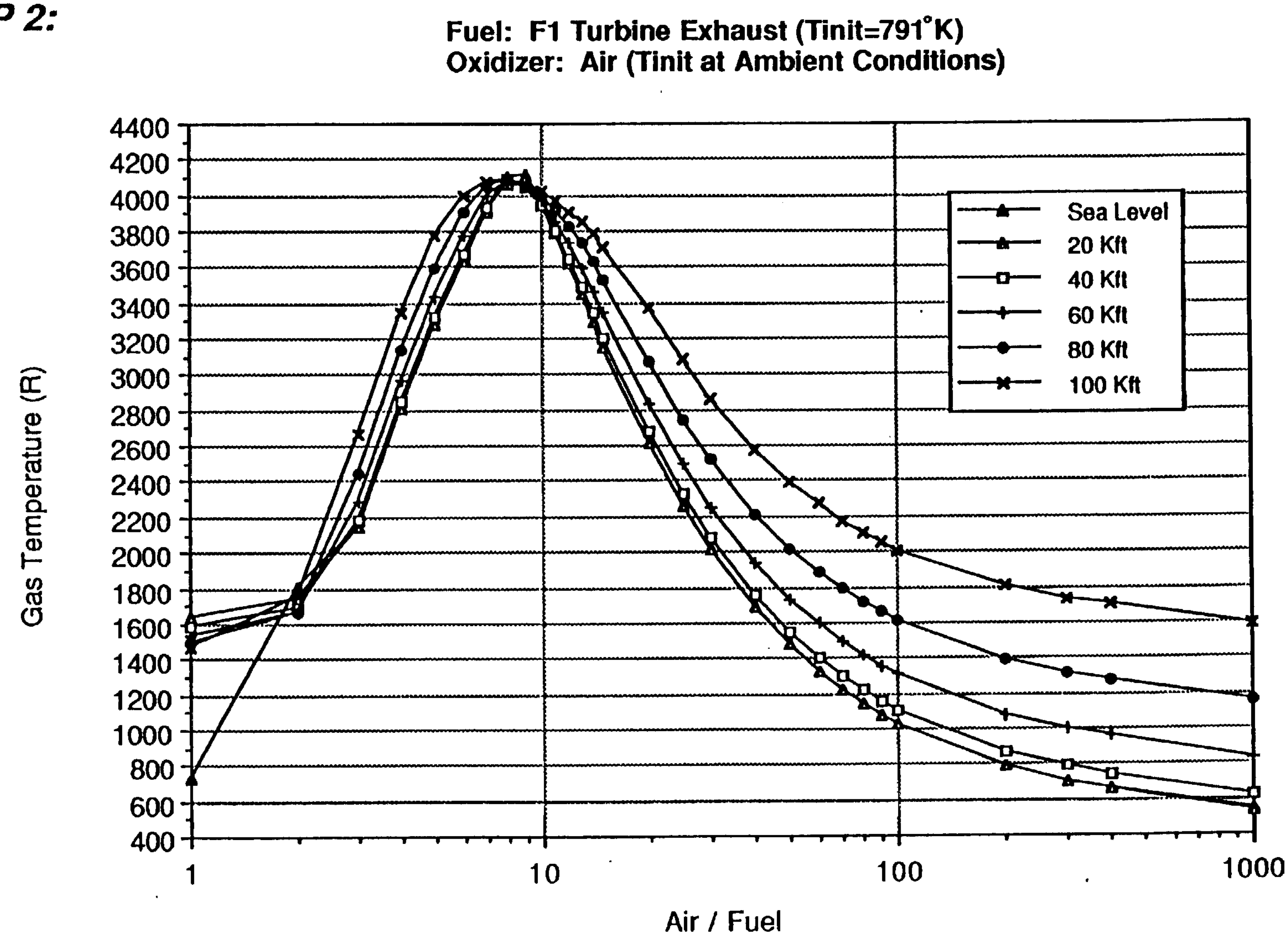
STEP 1:

Saturn V (S1-C) Base Heat Shield



COMBUSTION TEMPERATURES FOR F-1 ENGINE TURBINE EXHAUST WITH AIR

STEP 2:

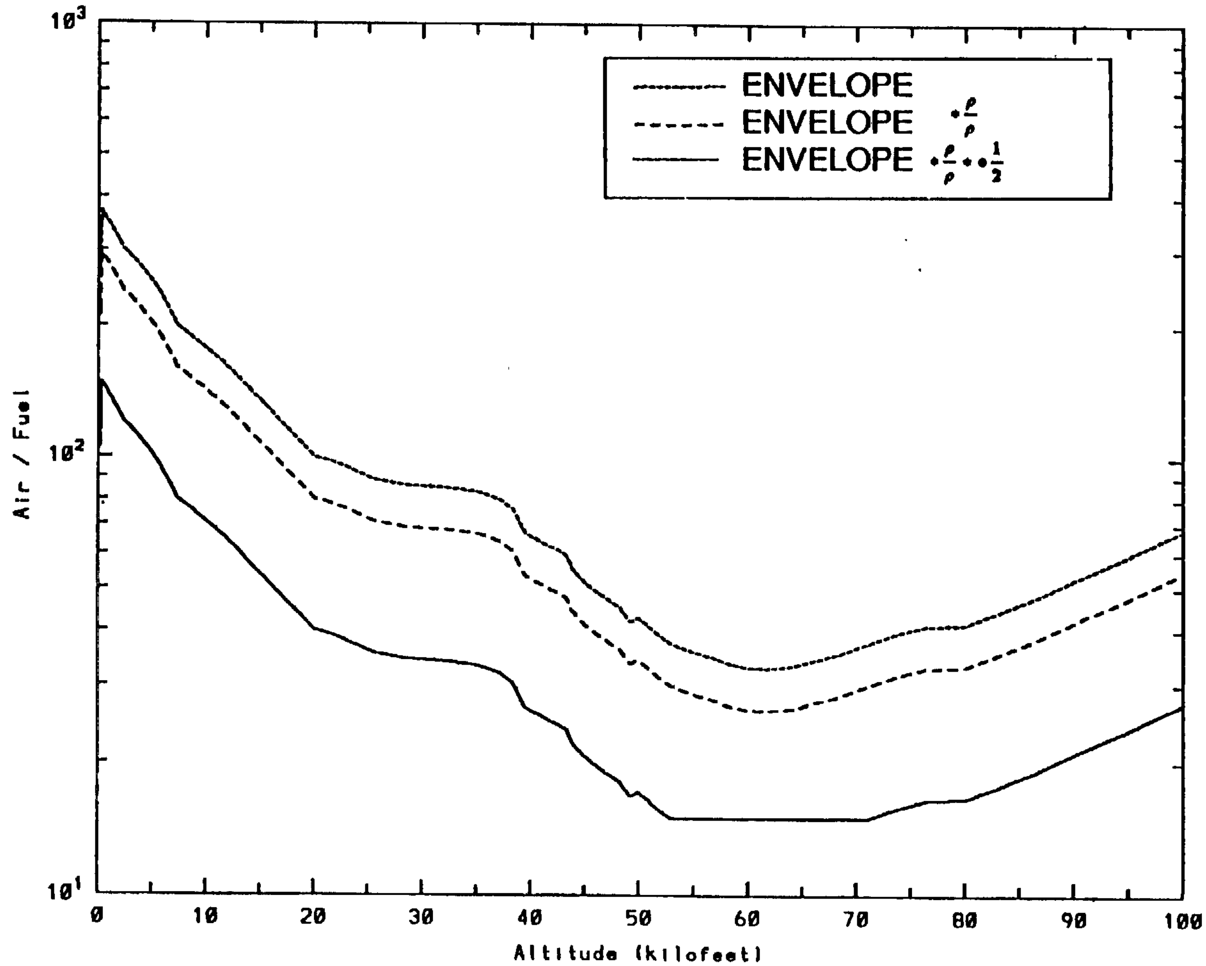


SATURN V BASE REGION AIR/FUEL RATIO CORRECTED TO NLS FLIGHT CONDITIONS

STEP 3:

Saturn V (S1-C) Base Heat Shield

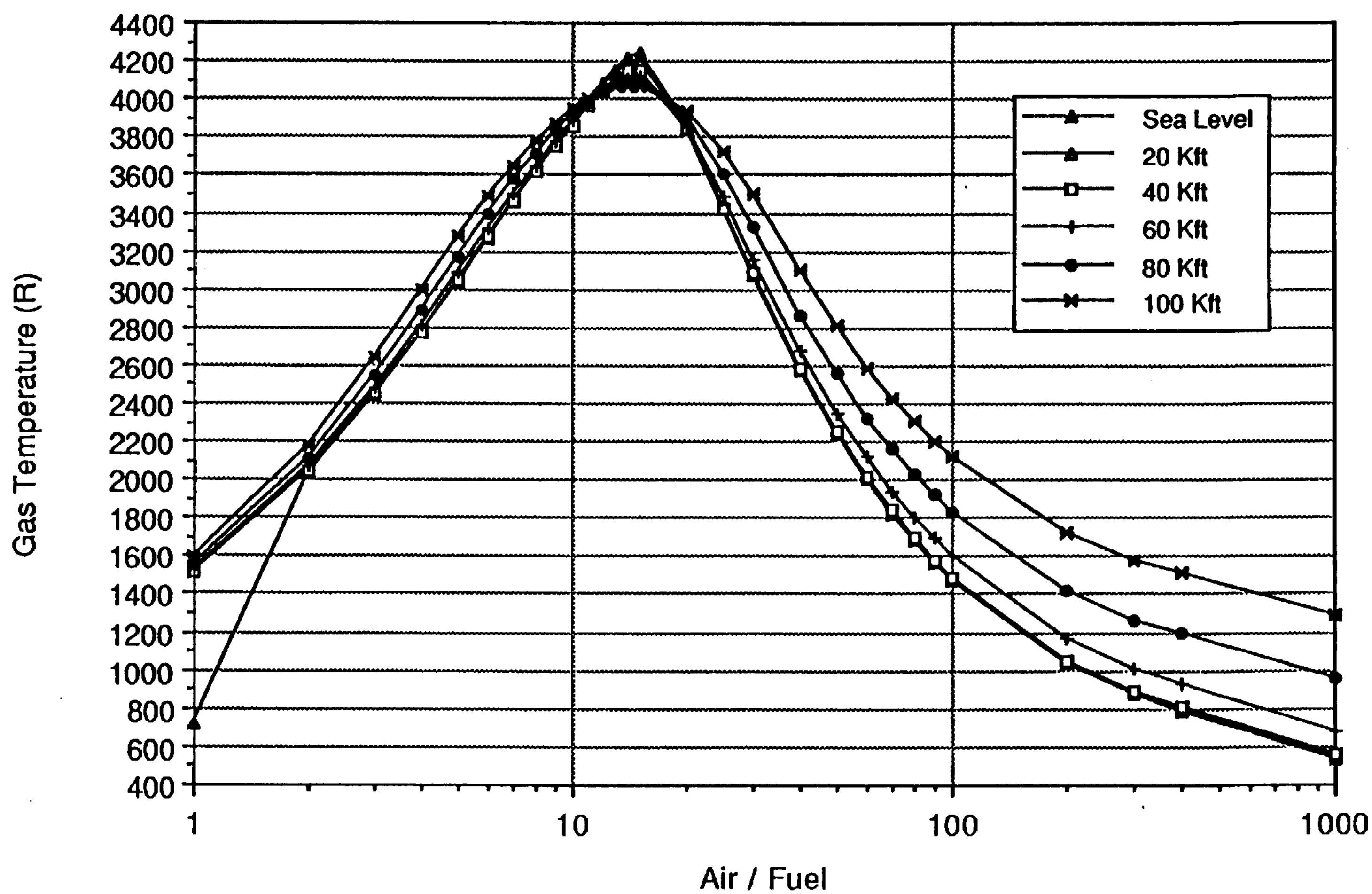
F-1 Air / Fuel Ratios



COMBUSTION TEMPERATURES FOR STME TURBINE EXHAUST WITH AIR

STEP 4:

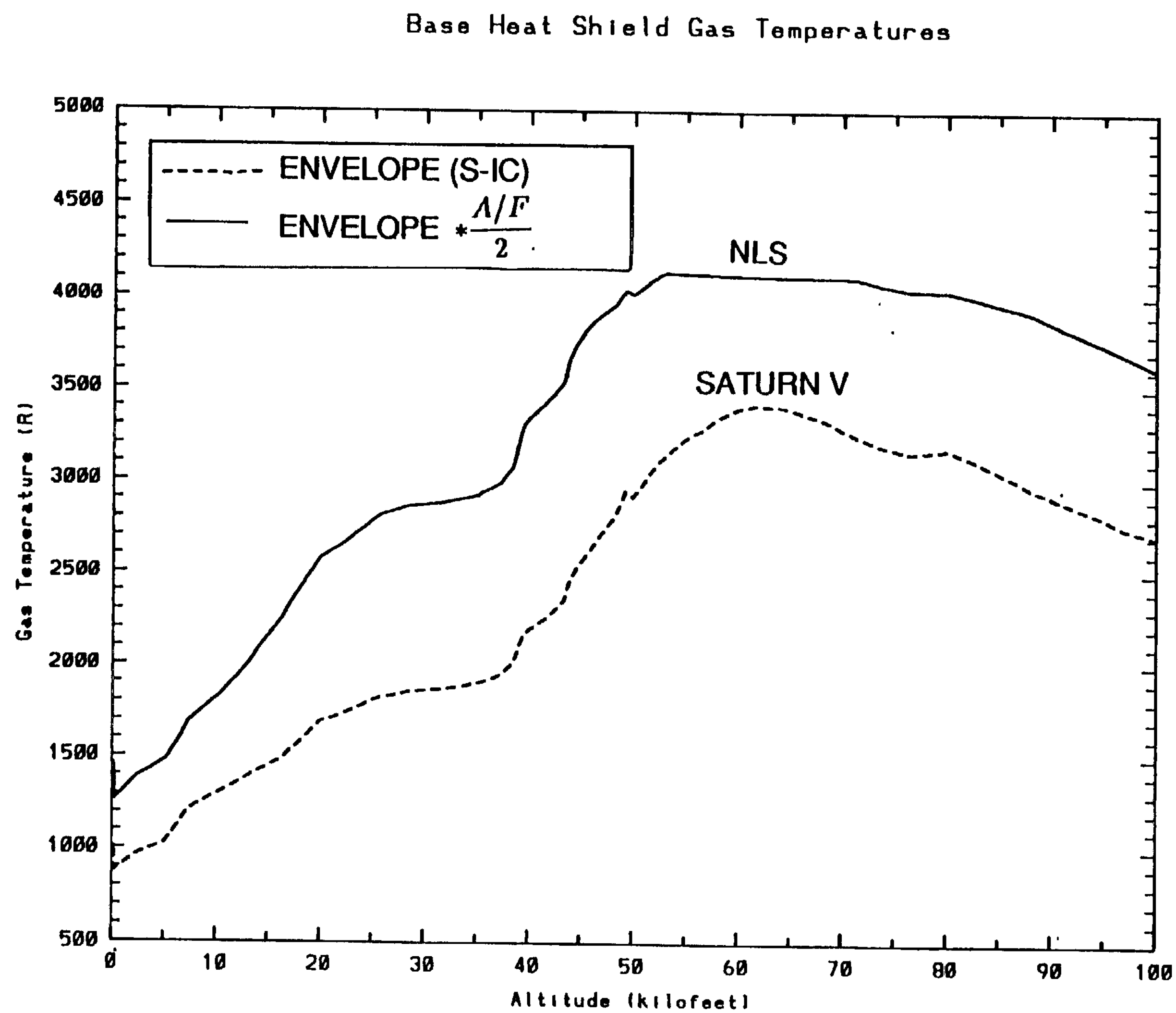
Fuel: H2 & H2O (Tinit = 460°K)
Oxidizer: Air (Tinit at Ambient Conditions)



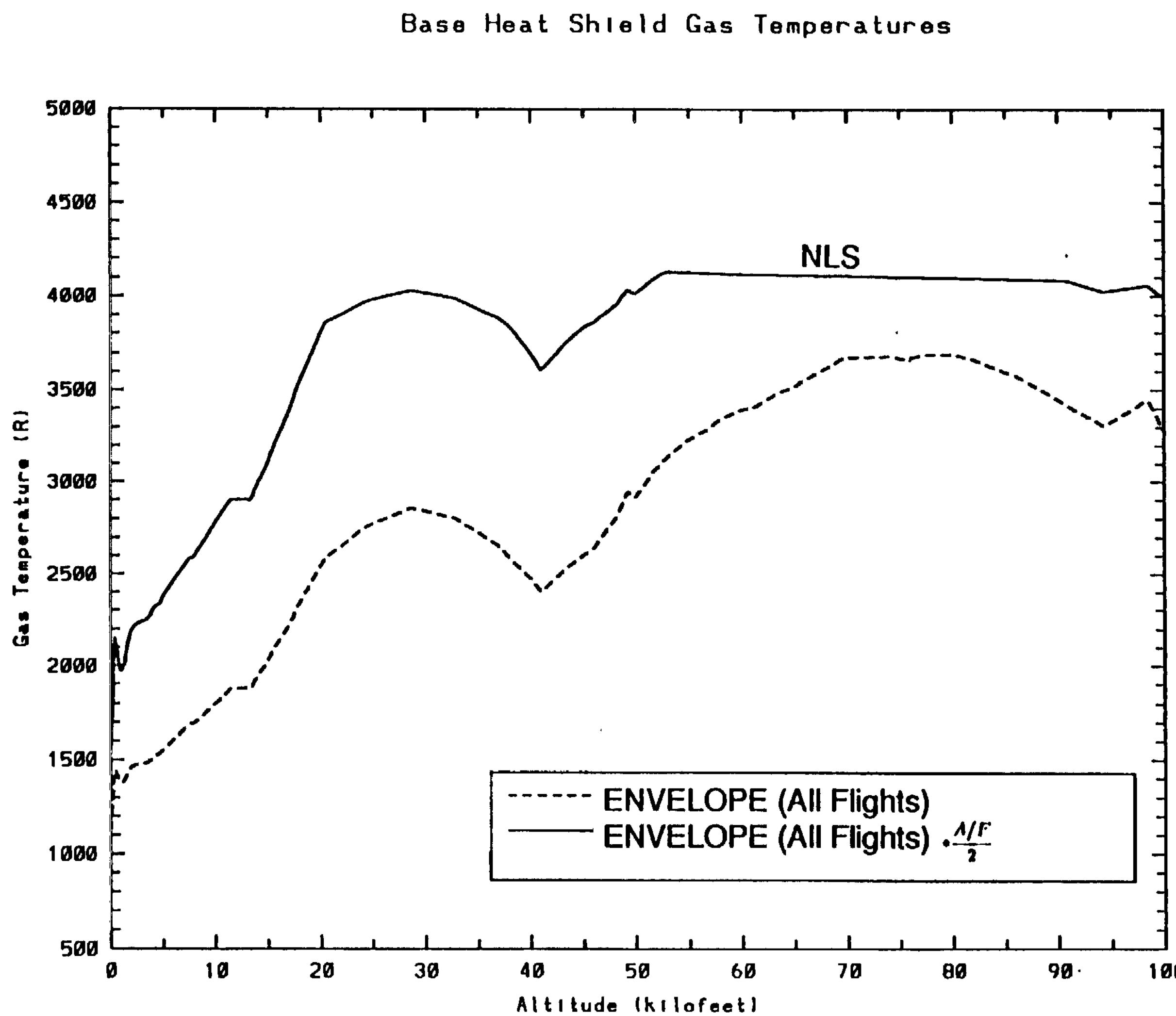
NLS 1.5 STAGE BASE HEAT SHIELD GAS TEMPERATURE ESTIMATES DEVELOPED FROM SATURN V FLIGHT DATA

REMTECH

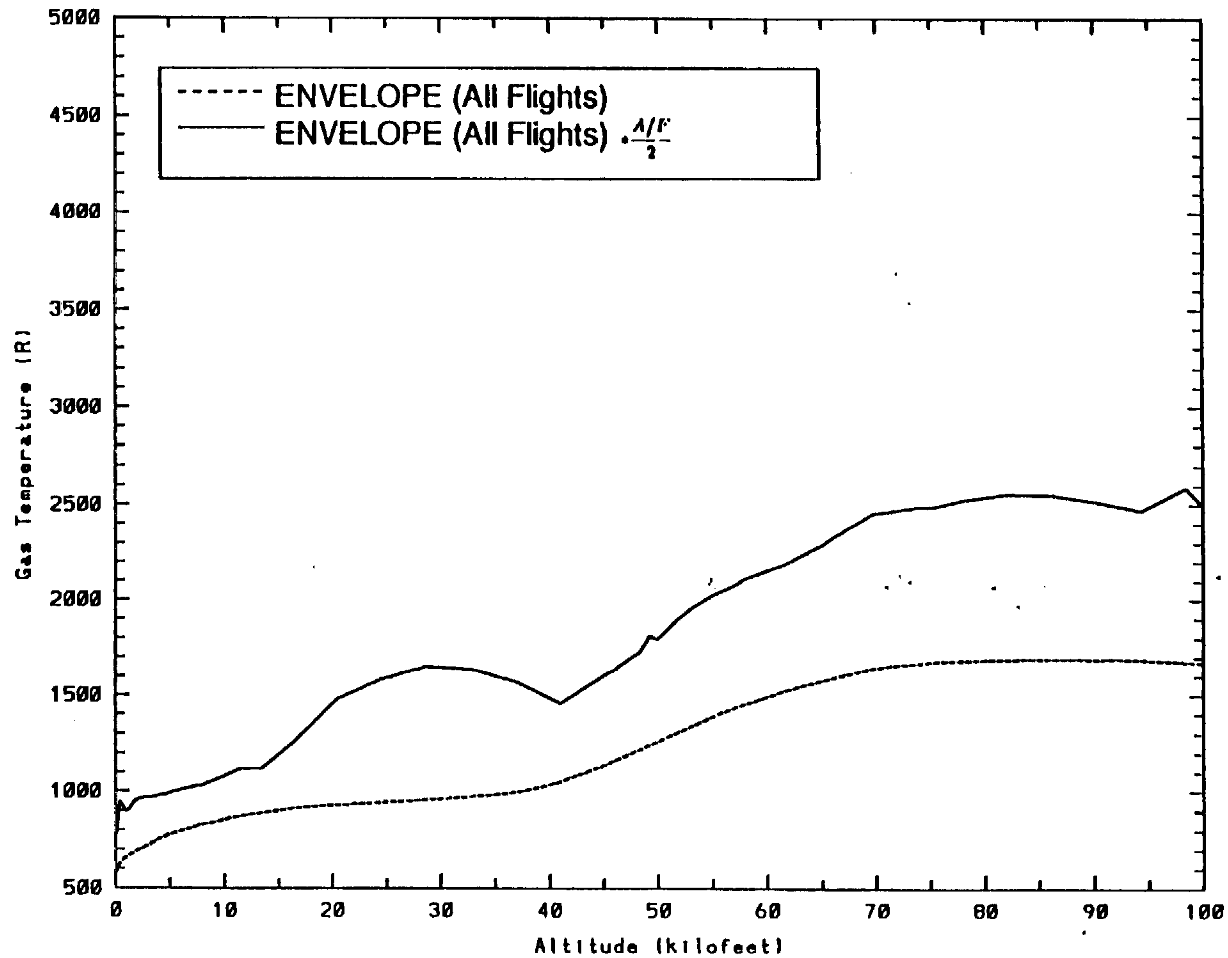
STEP 5:



NLS 1.5 STAGE BASE HEAT SHIELD GAS TEMPERATURE ESTIMATES DEVELOPED FROM ALL SATURN FLIGHT DATA

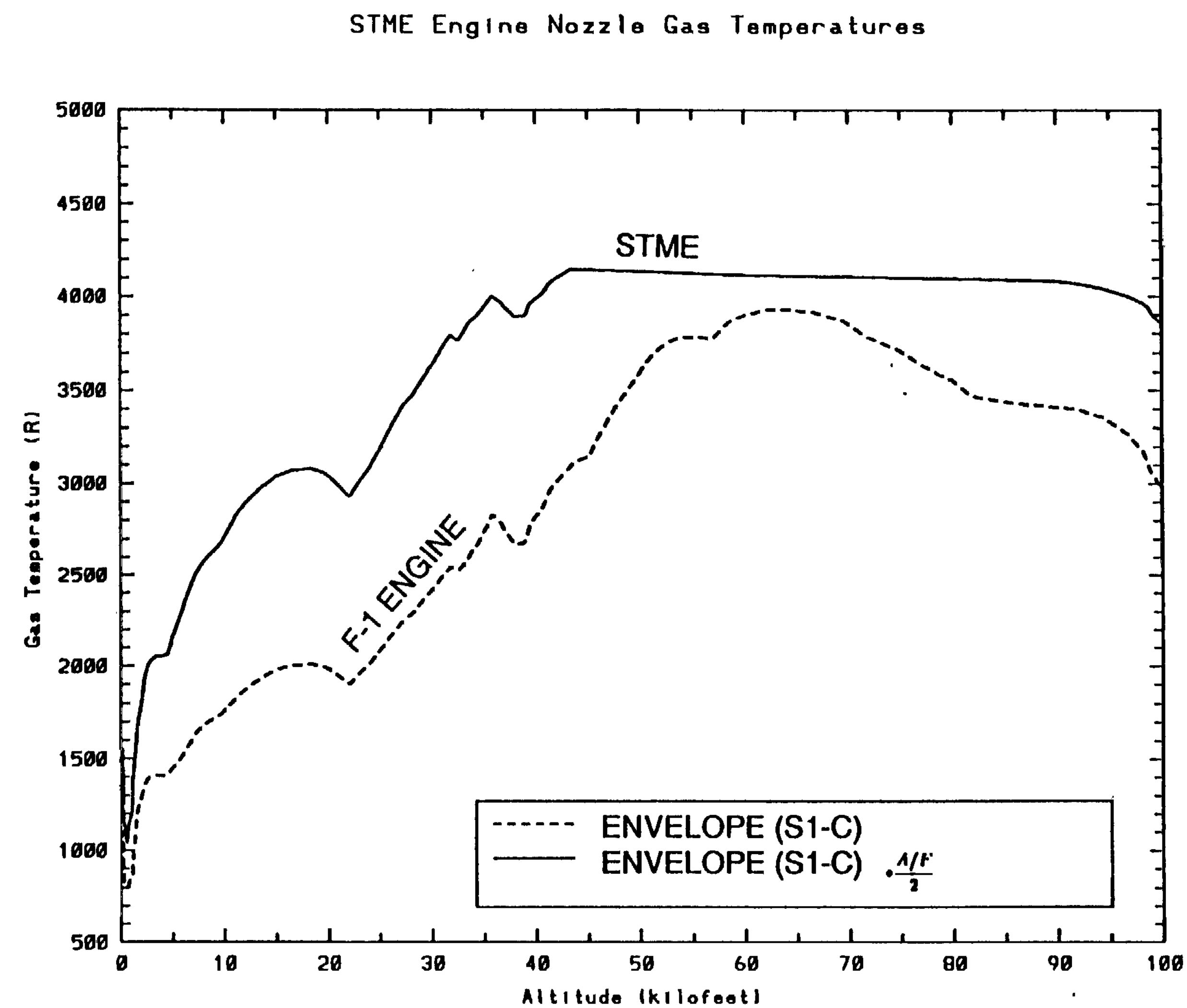


SATURN V S-1C STAGE F-1 ENGINE FLIGHT TEMPERATURE HISTORIES

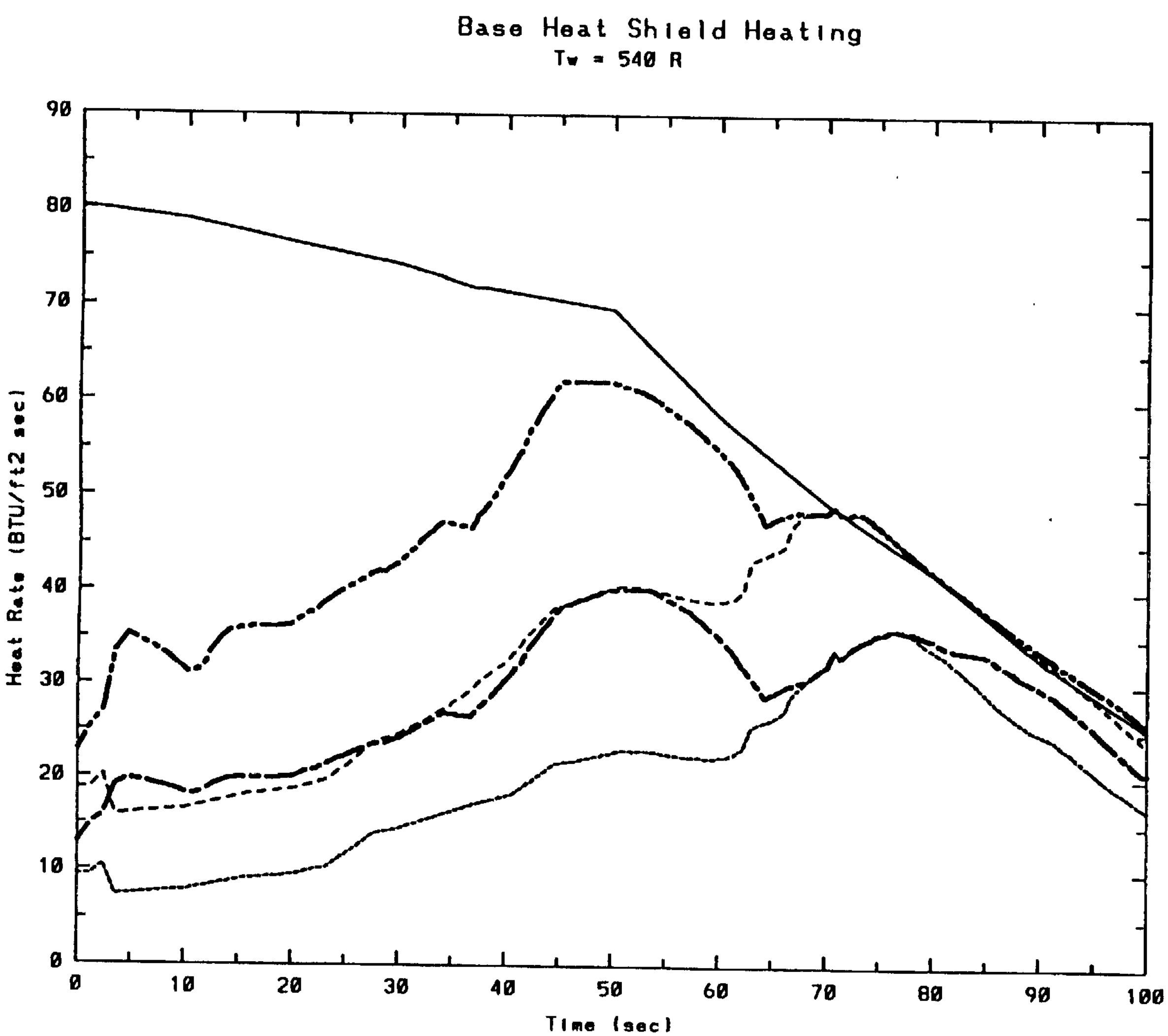


NLS 1.5 STAGE STME NOZZLE GAS TEMPERATURE ESTIMATES DEVELOPED FROM SATURN V FLIGHT DATA

REMTECH



NLS 1.5 STAGE BASE HEAT SHIELD CONVECTIVE HEATING RATES - $T_{WALL} = 540^{o}R$

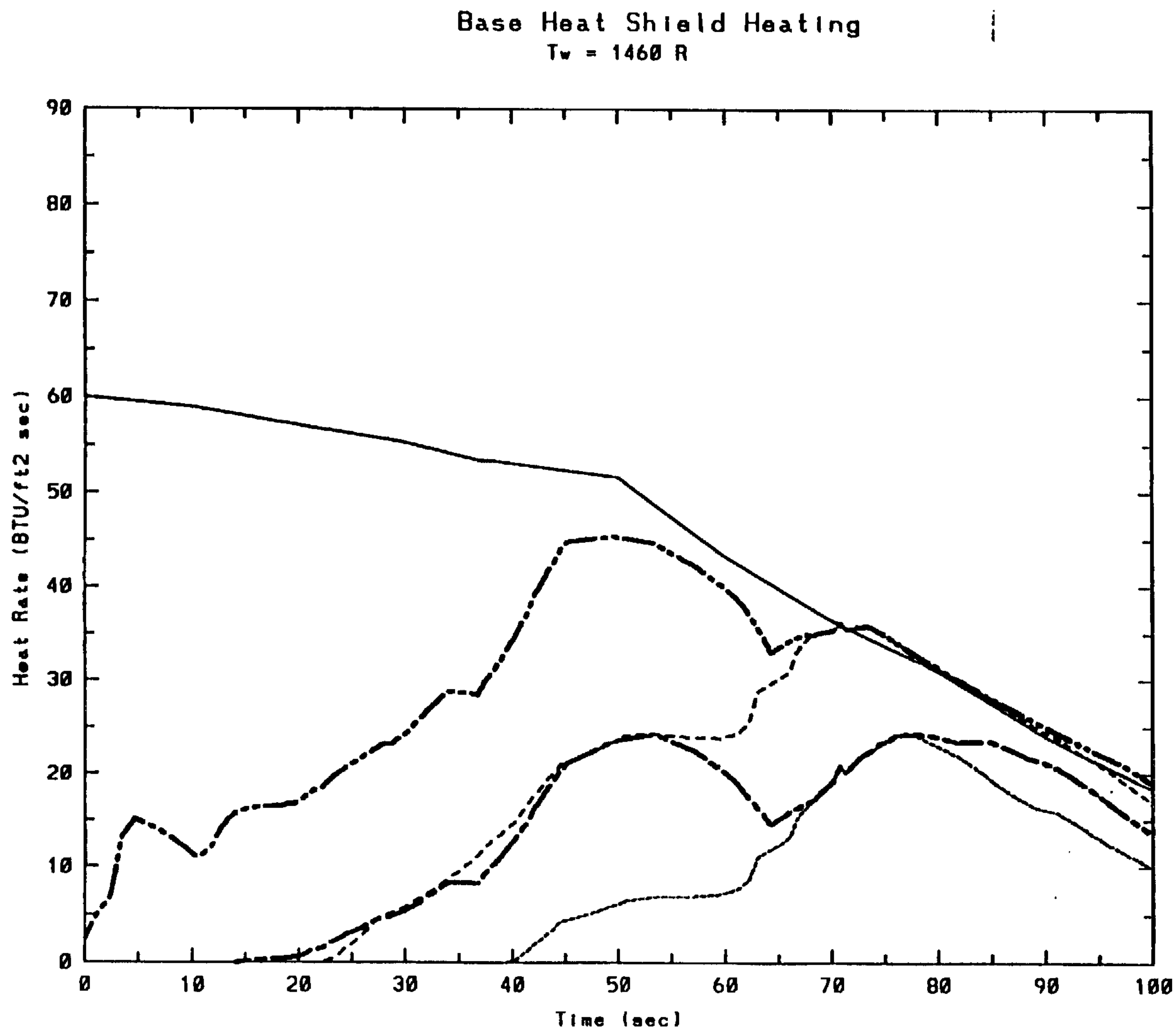


	$\dot{q}_{max}$ $(BTU/ft^2/sec)$	Q (BTU/ft^2)
Cycle 1 Design	80.3	6123
Envelope (S1-C)	36.0	2049
Envelope (S1-C) $\cdot \frac{A/F}{2}$	49.0	3246
Envelope (All)	40.2	2917
Envelope (All) $\cdot \frac{A/F}{2}$	62.0	4427

GasTemperature Probe Flight Data

- S1-C — 22 Saturn V (S1-C)
- All — 19 Saturn I Block I
  42 Saturn I Block II
  22 Saturn V (S1-C)
  83 Total

NLS 1.5 STAGE BASE HEAT SHIELD CONVECTIVE HEATING RATES - $T_{WALL} = 1460^{o}$ R

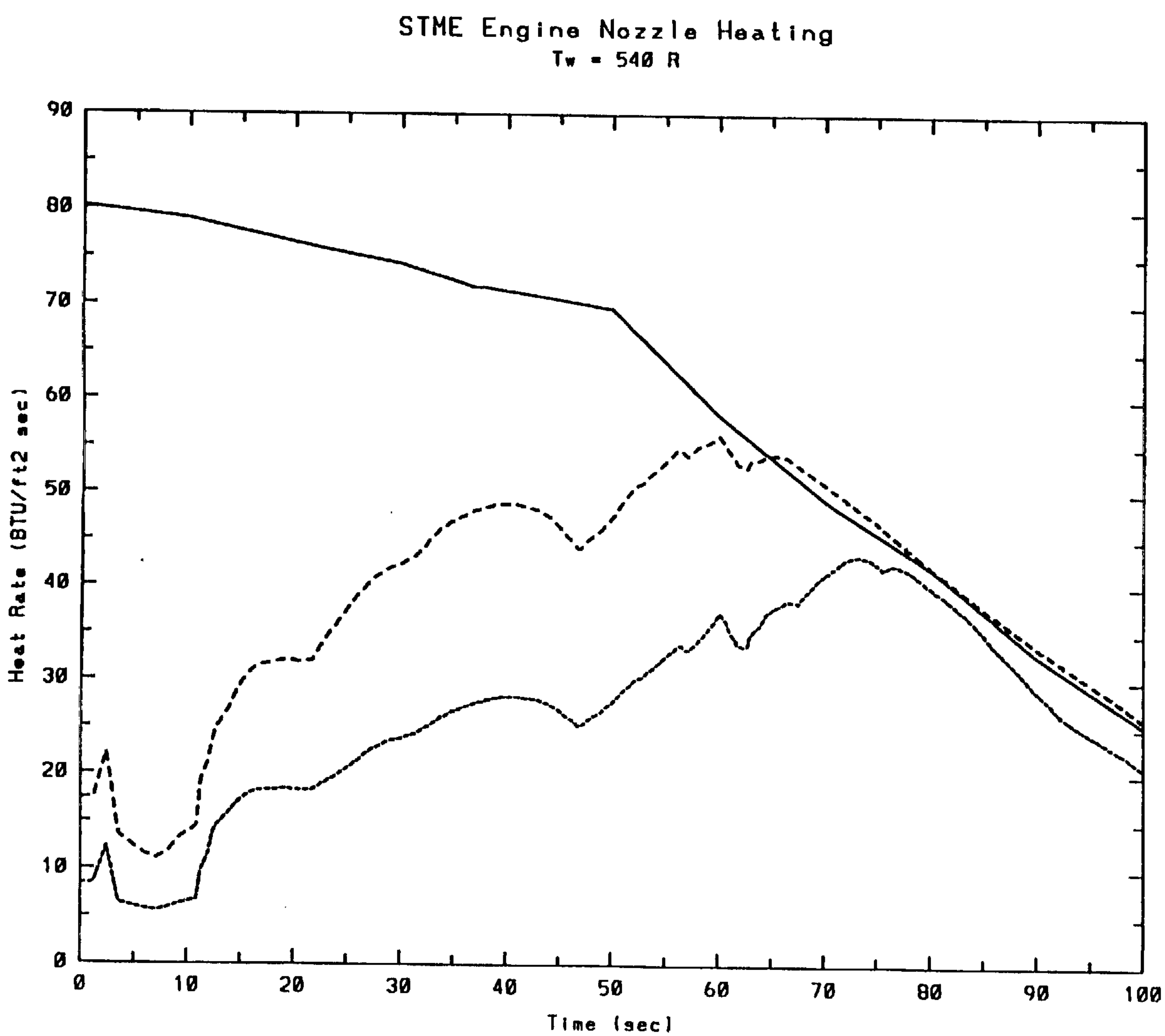


	$\dot{q}_{max}$ $(BTU/ft^2/sec)$	Q (BTU/ft^2)
Cycle 1 Design	60.1	4545
Envelope (S1-C)	24.4	494
Envelope (S1-C) $* \frac{A/F}{2}$	36.1	1691
Envelope (All)	24.4	1362
Envelope (All) $* \frac{A/F}{2}$	45.4	2872

Gas Temperature Probe Flight Data

• S1-C	22 Saturn V (S1-C)
• All	19 Saturn I Block I
	42 Saturn I Block II
	22 Saturn V (S1-C)
	83 Total

NLS 1.5 STAGE STME NOZZLE CONVECTIVE HEATING RATE - $T_{WALL} = 540^{o}$ R



	$\dot{q}_{max}$ $(BTU/ft^2/sec)$	Q (BTU/ft^2)
—— Cycle 1 Design	80.3	6123
······ Envelope (S1-C)	43.3	2730
- - - Envelope (S1-C) $* \frac{A/F}{2}$	56.2	3990

GasTemperature Probe Flight Data

- S1-C   8 Saturn V (S1-C)

CONCLUSIONS

REMTECH

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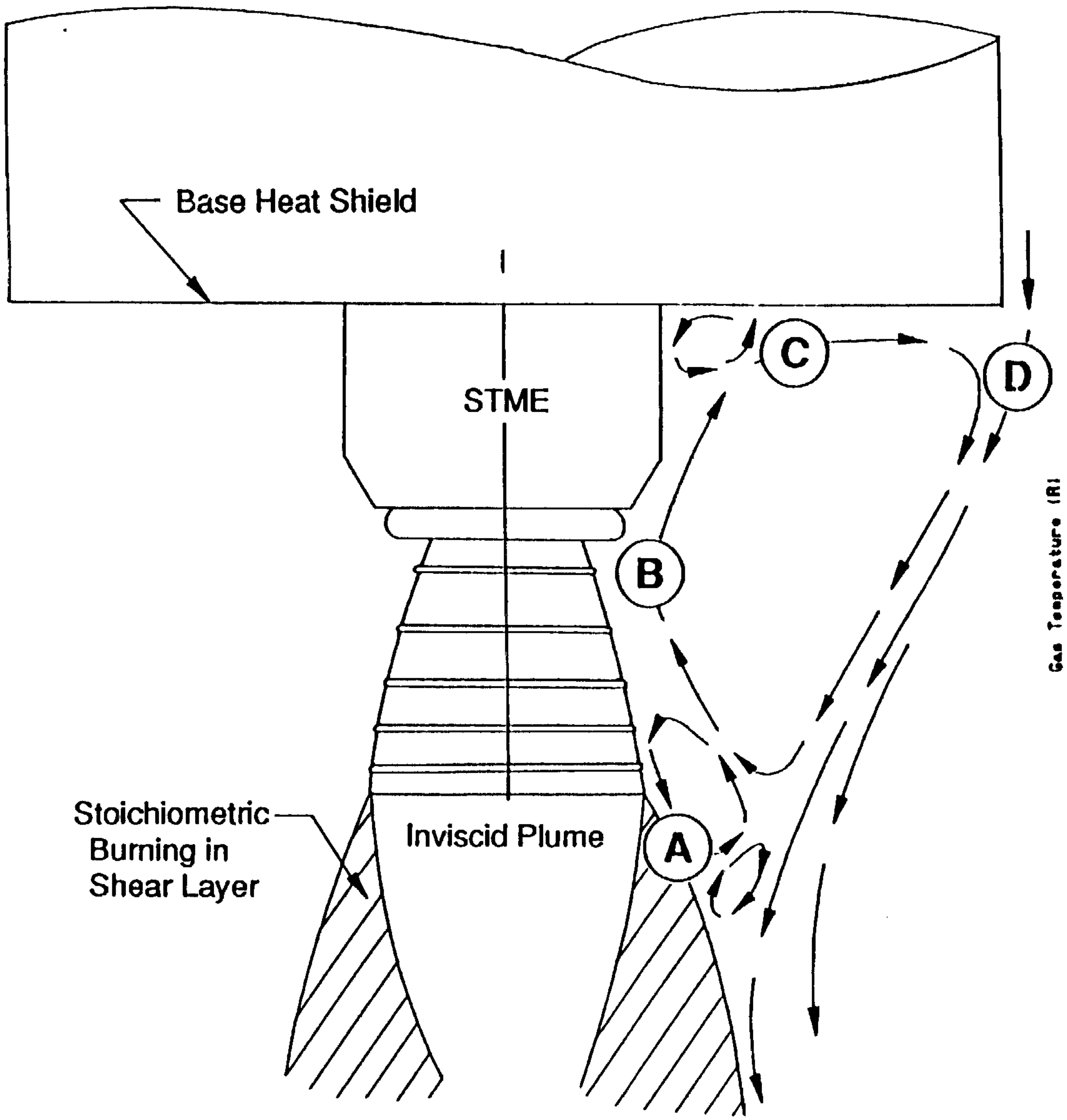
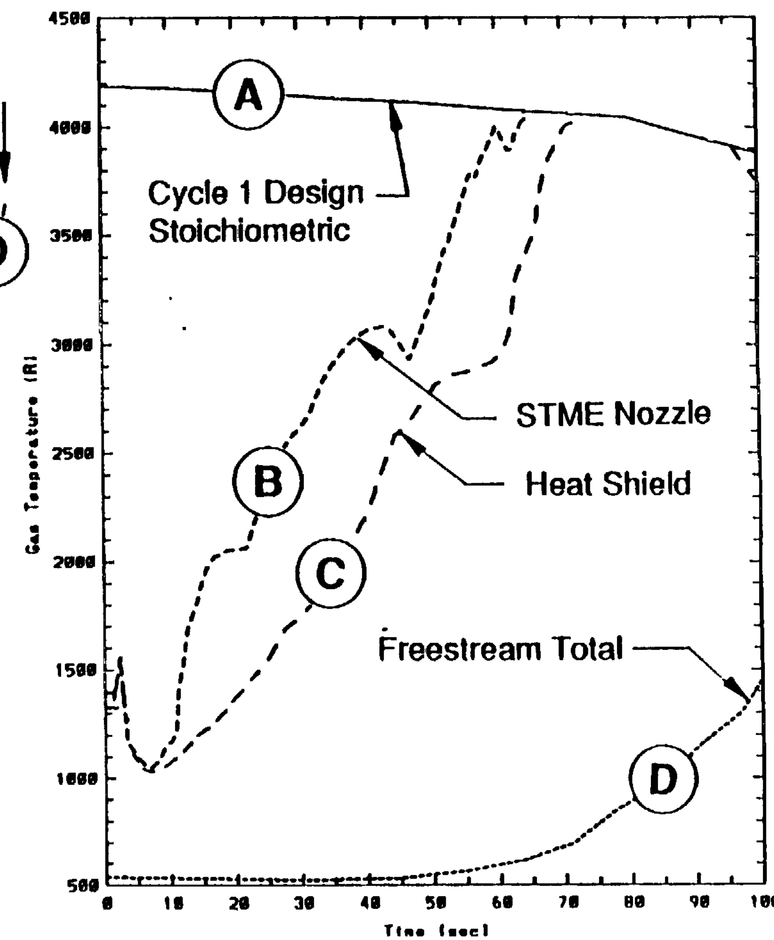
NLS SIMPLIFIED BASE REGION FLOWFIELD AT LOW ALTITUDES





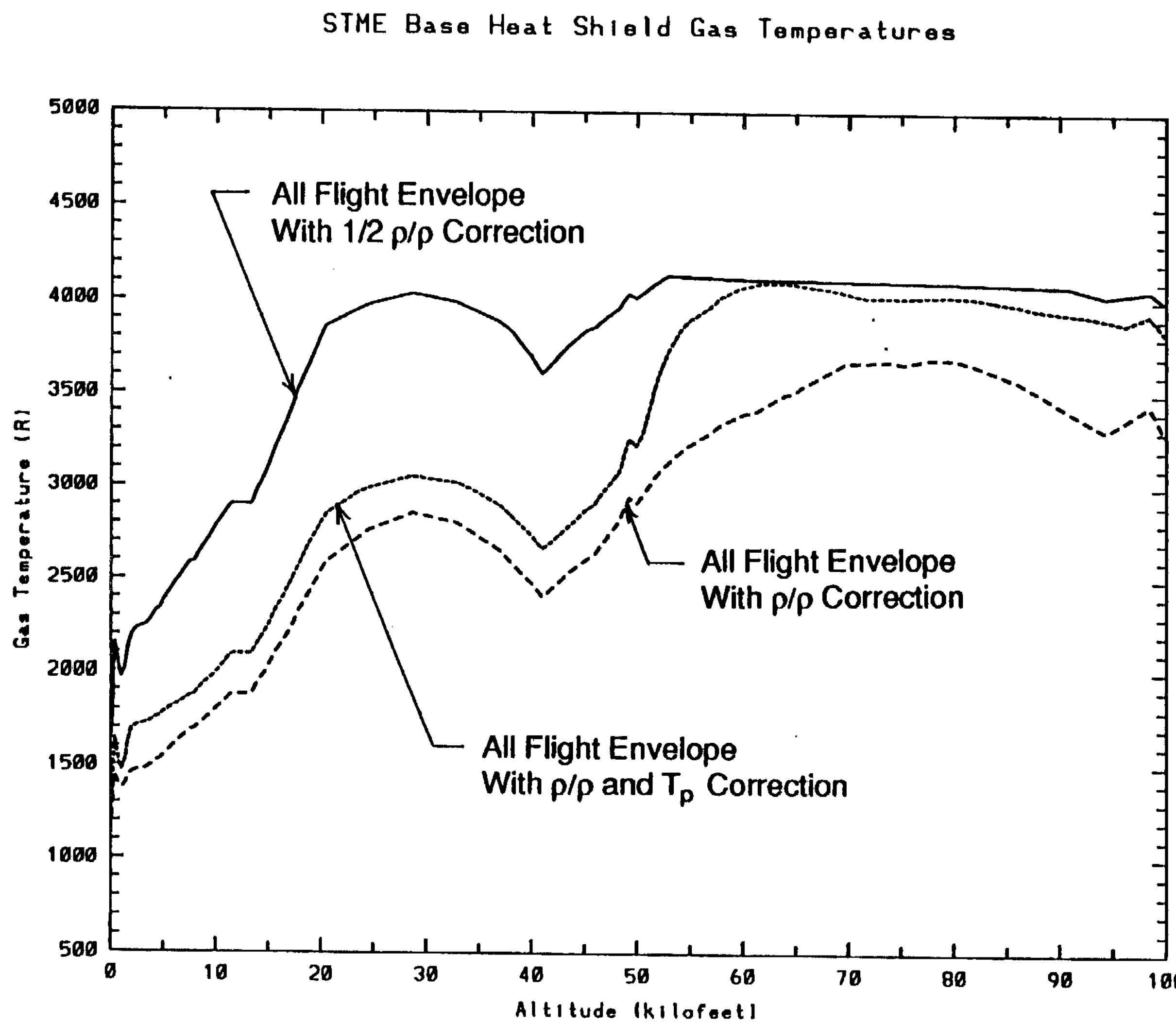
CONCLUSIONS

- Saturn flight base heating data are not presently useful in verifying the NLS Cycle 1 design convective heat transfer coefficient at altitudes below 35,000 feet.
- The Cycle 1 design coefficient (or a base region Reynolds number adjustment to the Cycle 1 design) should continue to be utilized for TPS studies.
- Saturn flight gas temperatures were less than stoichiometric burning levels (for air and F-1 or H-1 turbine exhaust) but greater than freestream total temperatures, early in flight.
- Air/turbine exhaust mixture ratios (in the base region) based upon measured gas temperatures can be adjusted to NLS 1.5 Stage conditions to obtain a ***reasonable*** upper limit estimate of NLS base gas recovery temperatures.
- NLS convective heating rates based upon the existing Cycle 1 design coefficient and improved gas temperature are approximately 50% lower than the Cycle 1 design environments.

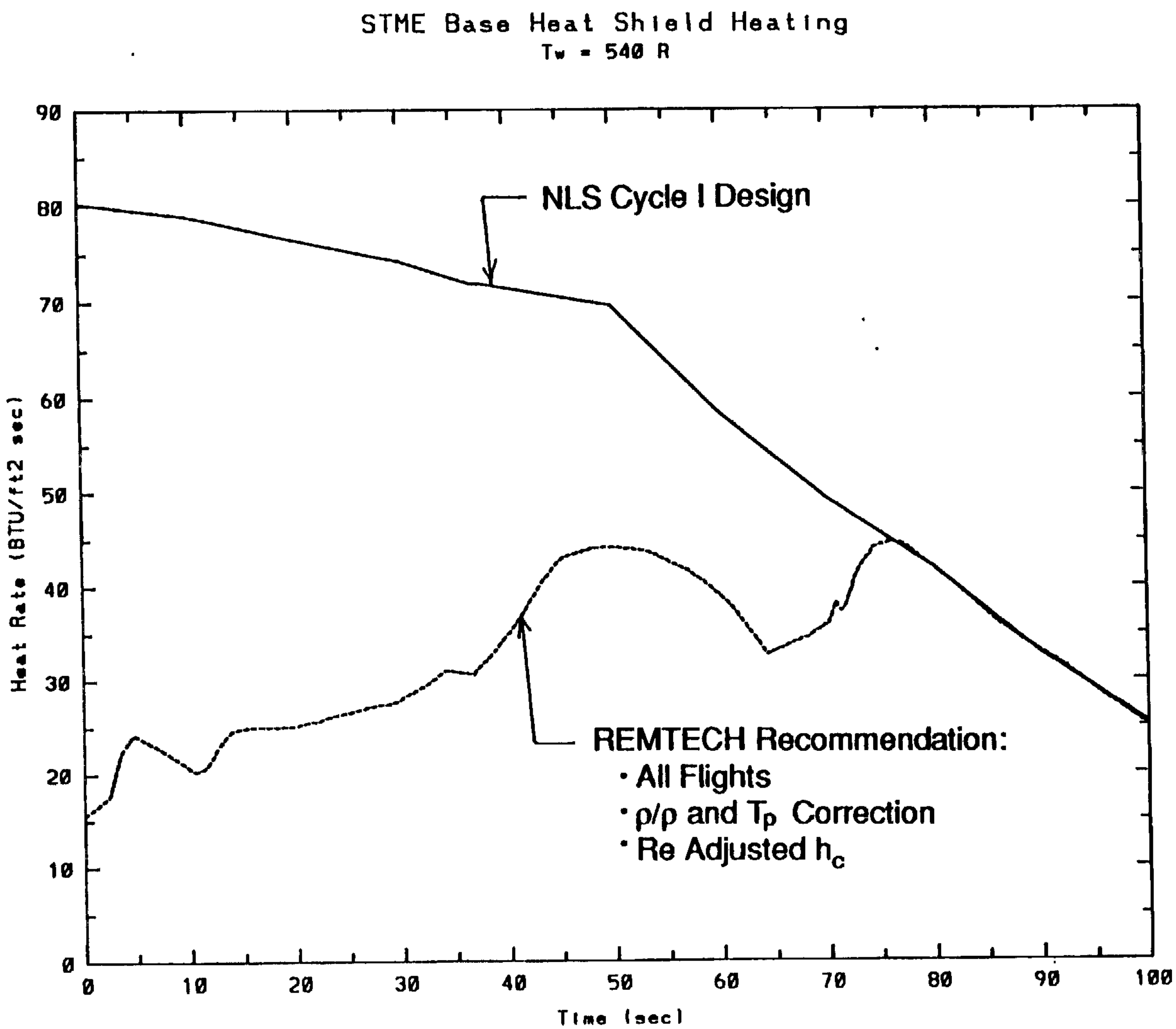
MATRIX OF CONVECTIVE BASE HEATING ENVIRONMENT OPTIONS FOR NLS 1.5 STAGE

CONVECTIVE HEAT TRANSFER COEFFICIENT DECREASING ↓	BASE GAS TEMPERATURE INCREASING →				
	Mean from Saturn V Flight with $\frac{\rho}{\rho_{corr}}$	All Saturn Envelope with $\frac{\rho}{\rho_{corr}}$	All Saturn Envelope with $\rho/\rho_{corr} + T_p\ corr$	All Saturn Envelope with $\frac{1}{2}\ \frac{\rho}{\rho\ corr}$	Cycle 1 Stoichiometric Combustion
Cycle 1 Saturn I Block II Average from CR 61390	26.2 1270	40.2 2917	44.7 3363	62.0 4427	Cycle 1 Design 80.3 6123
Cycle 1 with Re adjustment from $t = 0$ to $t = 75$ sec	26.1 1078	35.8 2226	REMTECH Recommendation 44.4 2554	51.0 3232	
S-IC Design from Boeing FTS-H-174	22.3 998	32.4 2114	37.8 2427	49.8 3081	

NLS 1.5 STAGE BASE HEAT SHIELD GAS TEMPERATURES



REMTECH RECOMMENDED CONVECTIVE HEATING RATE FOR THE NLS 1.5 STAGE BASE HEAT SHIELD



REMTECH

NLS

CONVECTIVE BASE HEATING INVESTIGATION

IMPROVED METHODOLOGY SENSITIVITY STUDIES

JUNE 4, 1992

PREPARED BY:
ROBERT L. BENDER
REMTECH inc.
3304 WESTMILL DRIVE
HUNTSVILLE, AL 35805

IMPROVED METHODOLOGY SENSITIVITY STUDIES

IMPROVED METHODOLOGY

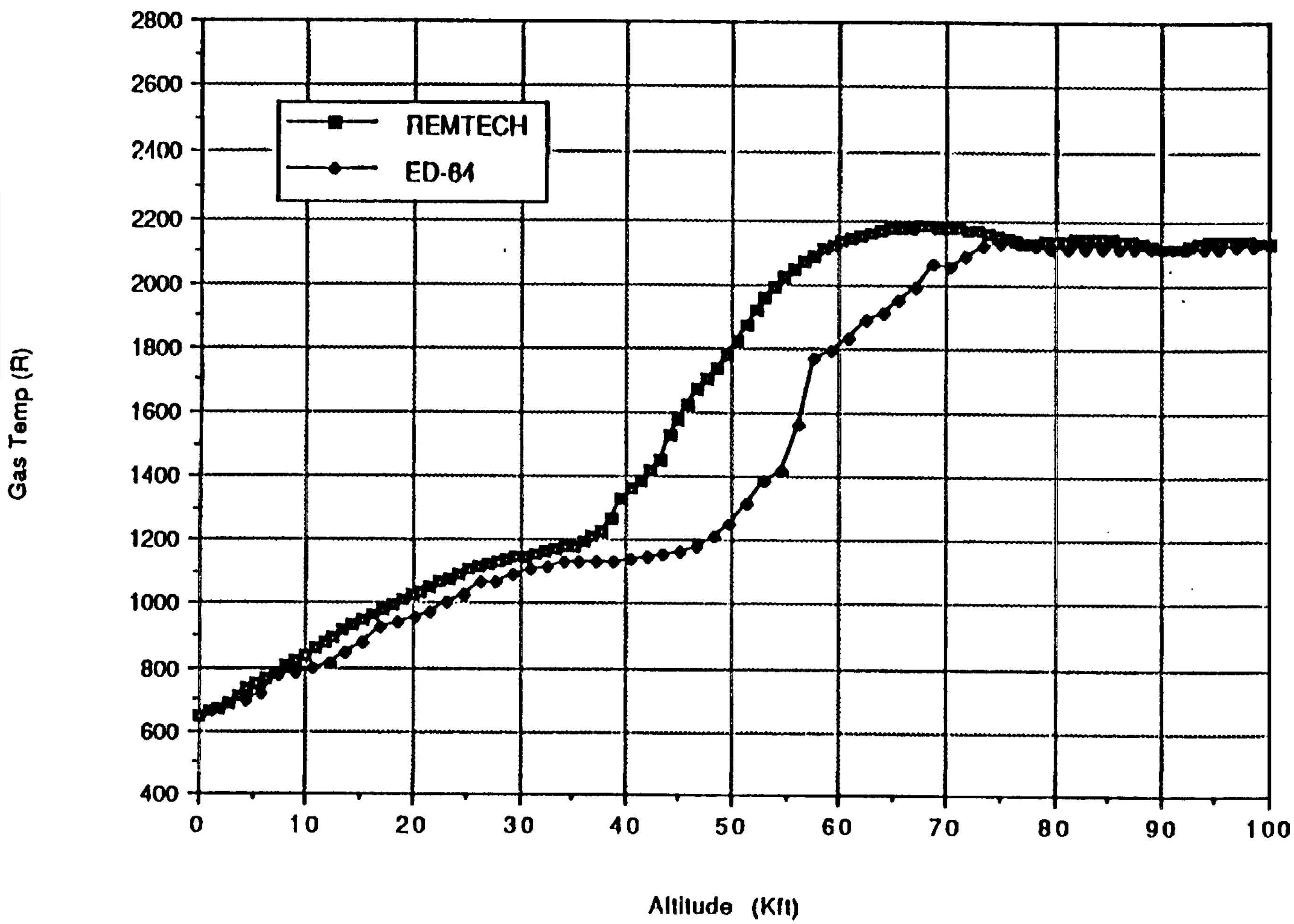
- Presented to ED Lab May 19, 1992
- Includes:
  - New methodology for h_c from 0 to 75 seconds (35,000 ft.)
  - Base gas temperature derived from Saturn flight experience

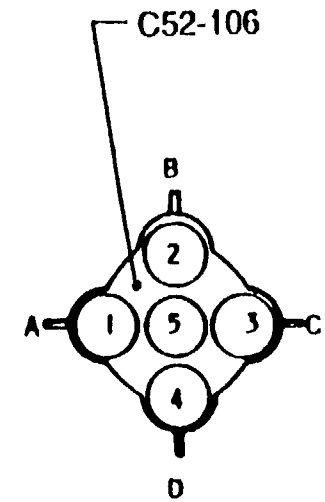
SENSITIVITY STUDIES AND METHODOLOGY VERIFICATION

- Comparison (validation) of database – REMTECH vs. ED-64
- Different methods for enveloping Saturn flight data
- Sensitivity of h_c and $\dot{q}_c$ to choice of base region velocity
- Sensitivity of T_{gas} to entrainment adjustment $\rho_{RP\text{-}1}/\rho H_2$
- Sensitivity of T_{gas} to F-1 engine exhaust – carbon burning
- Comparison of convective heat loads with different assumptions

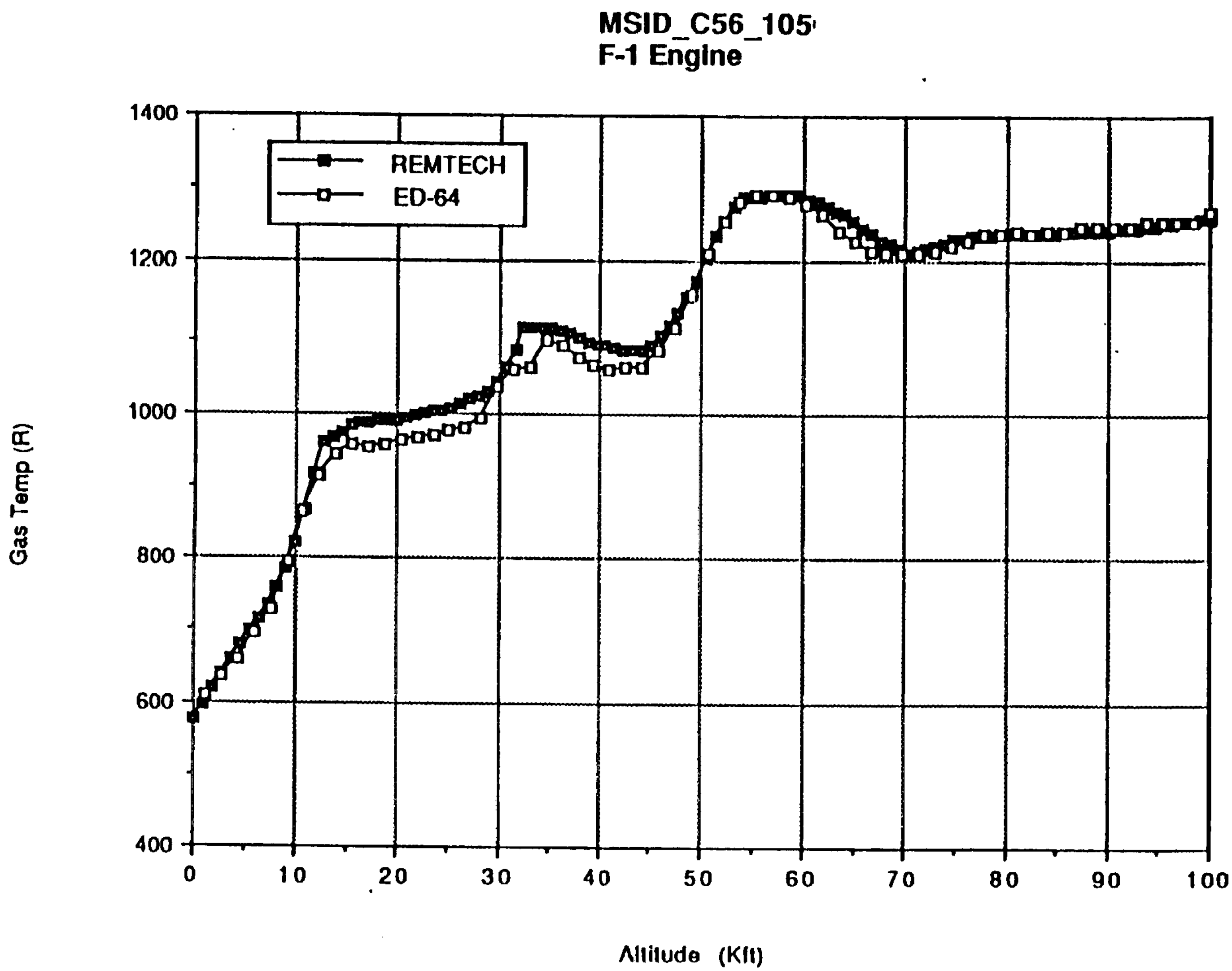
COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64

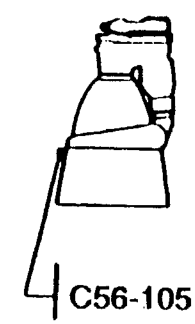
MSID_C52_106
Base Heat Shield



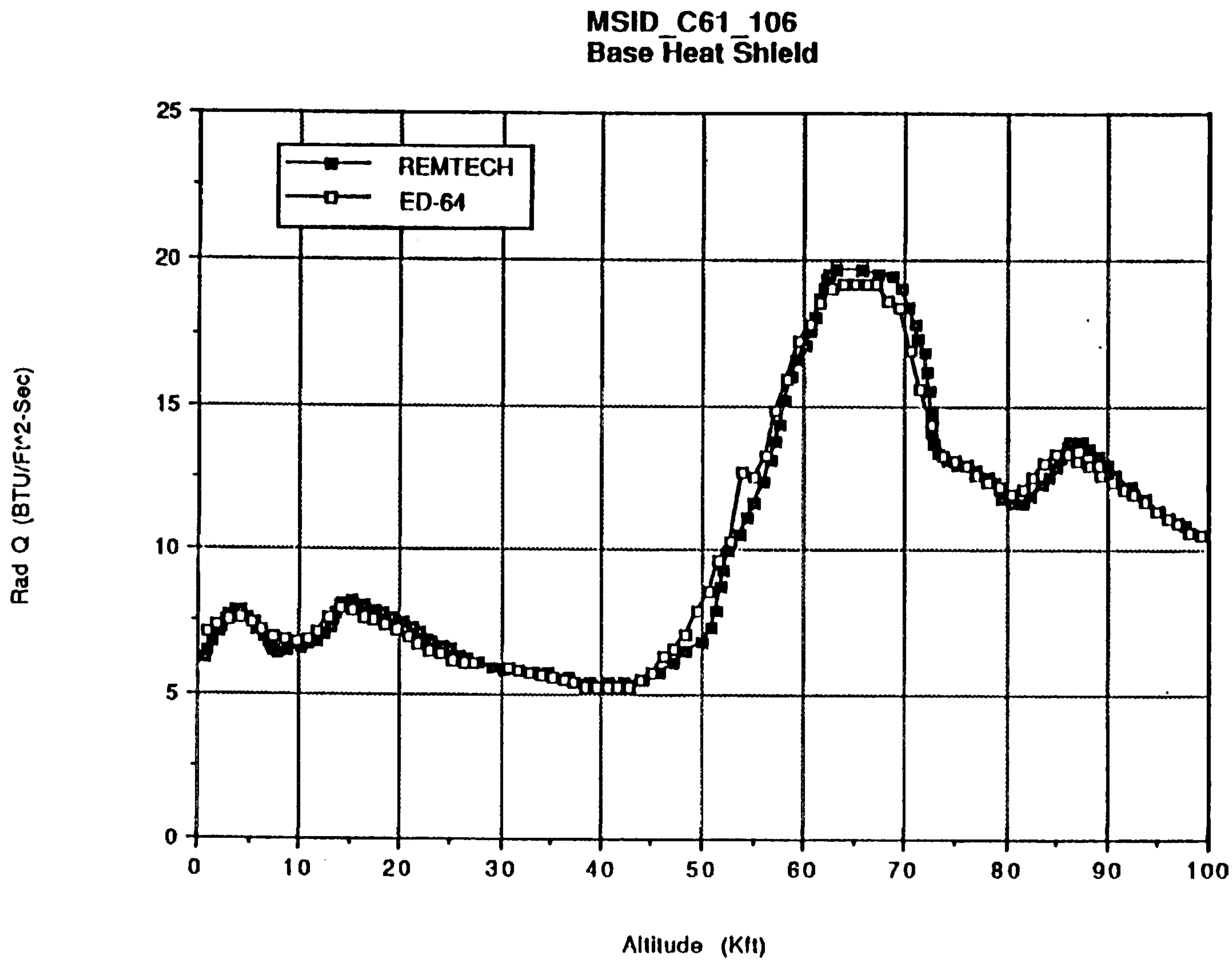


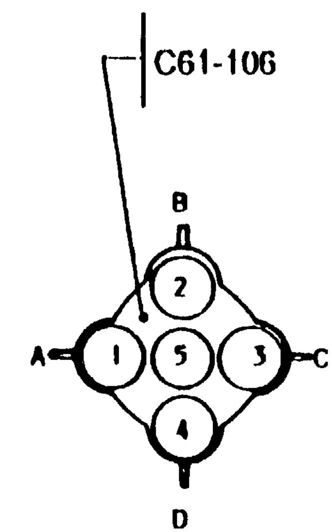
COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64



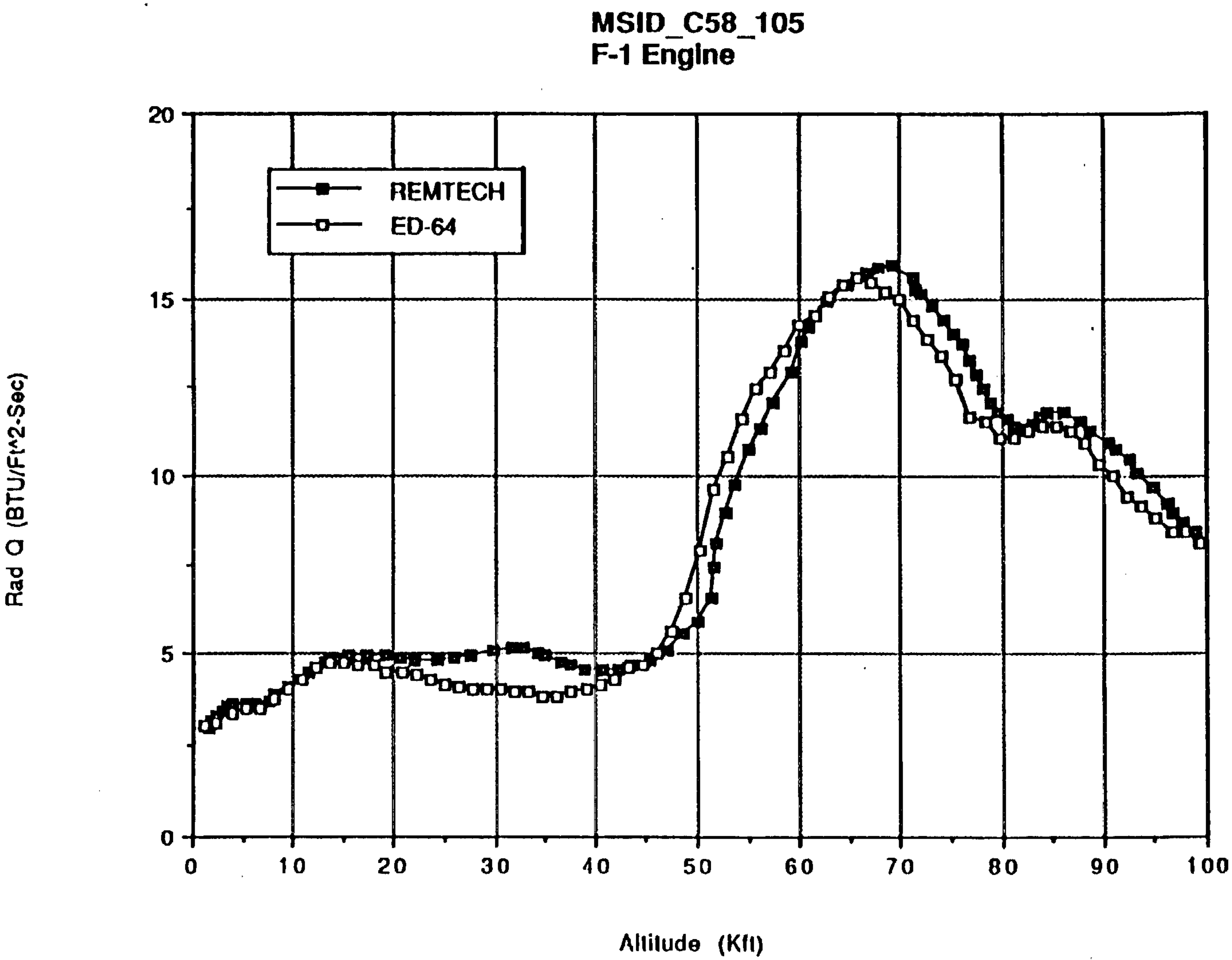


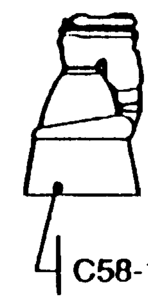
COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64





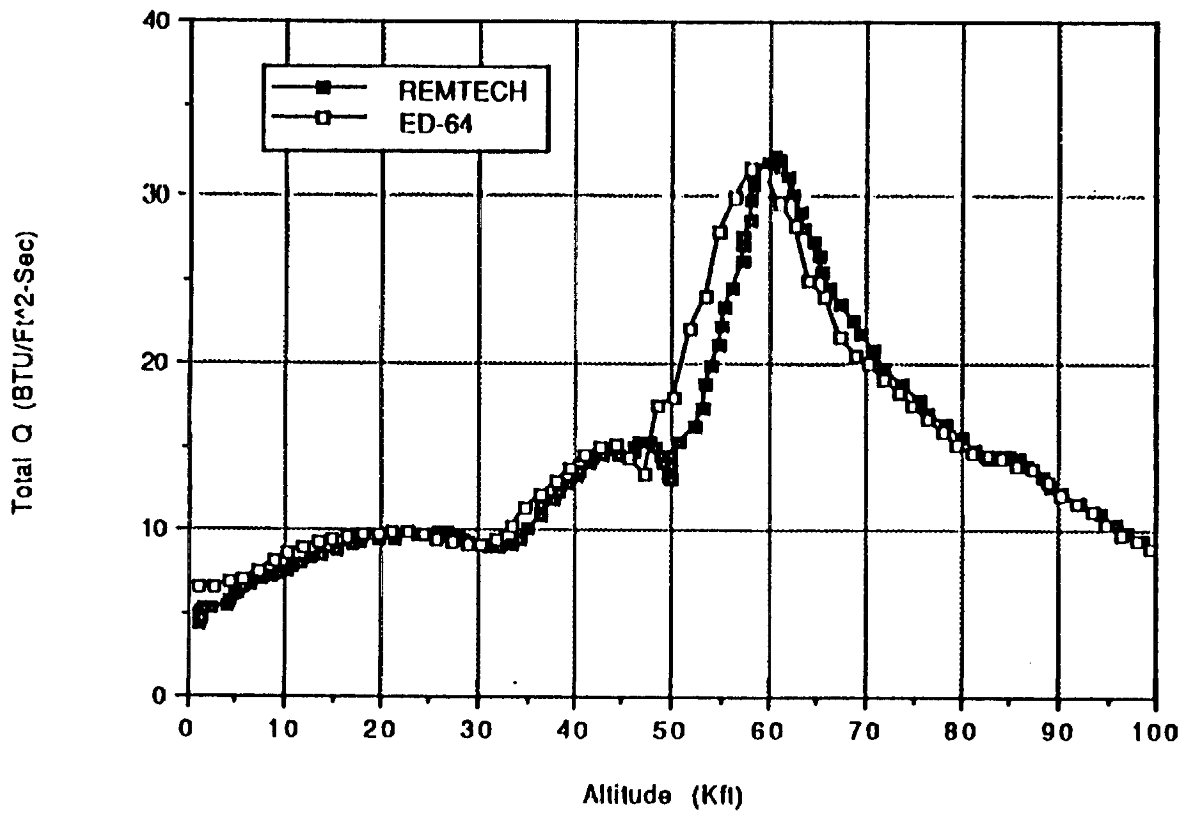
COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64

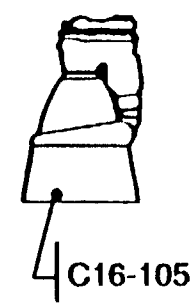




COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64

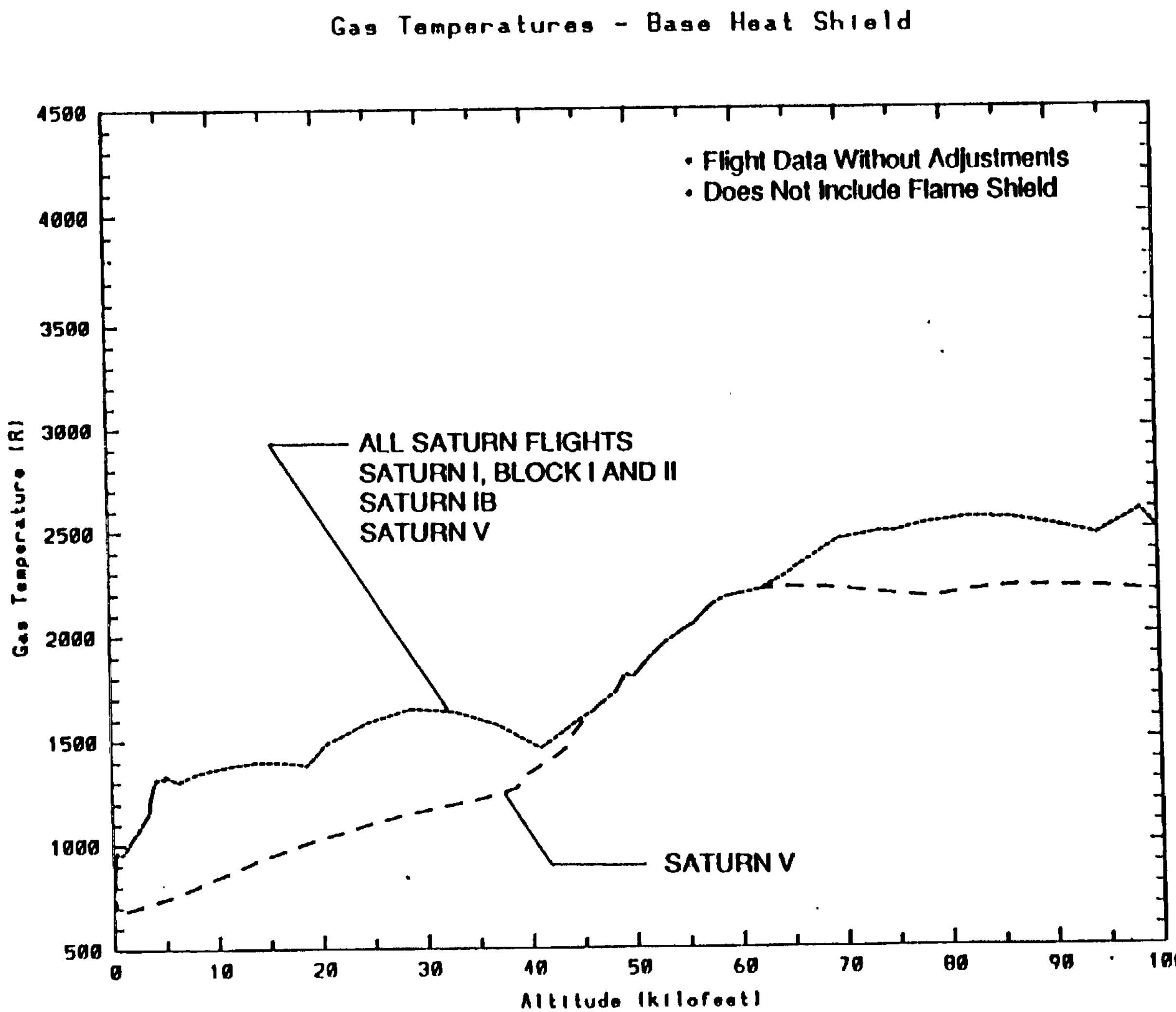
MSID_C16_105
F-1 Engine



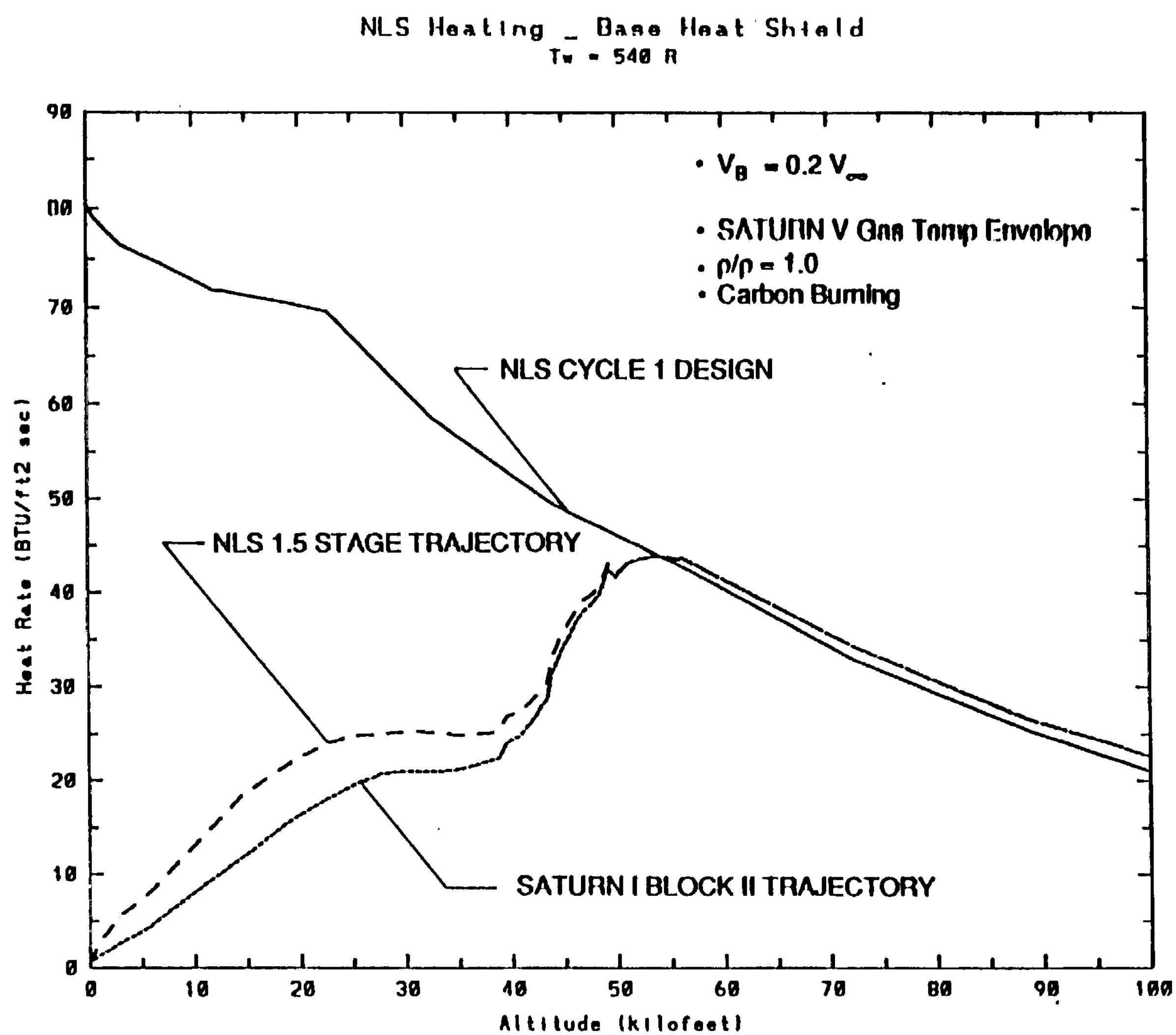


ENVELOPE OF FLIGHT GAS TEMPERATURES
SATURN V vs ALL SATURN FLIGHTS

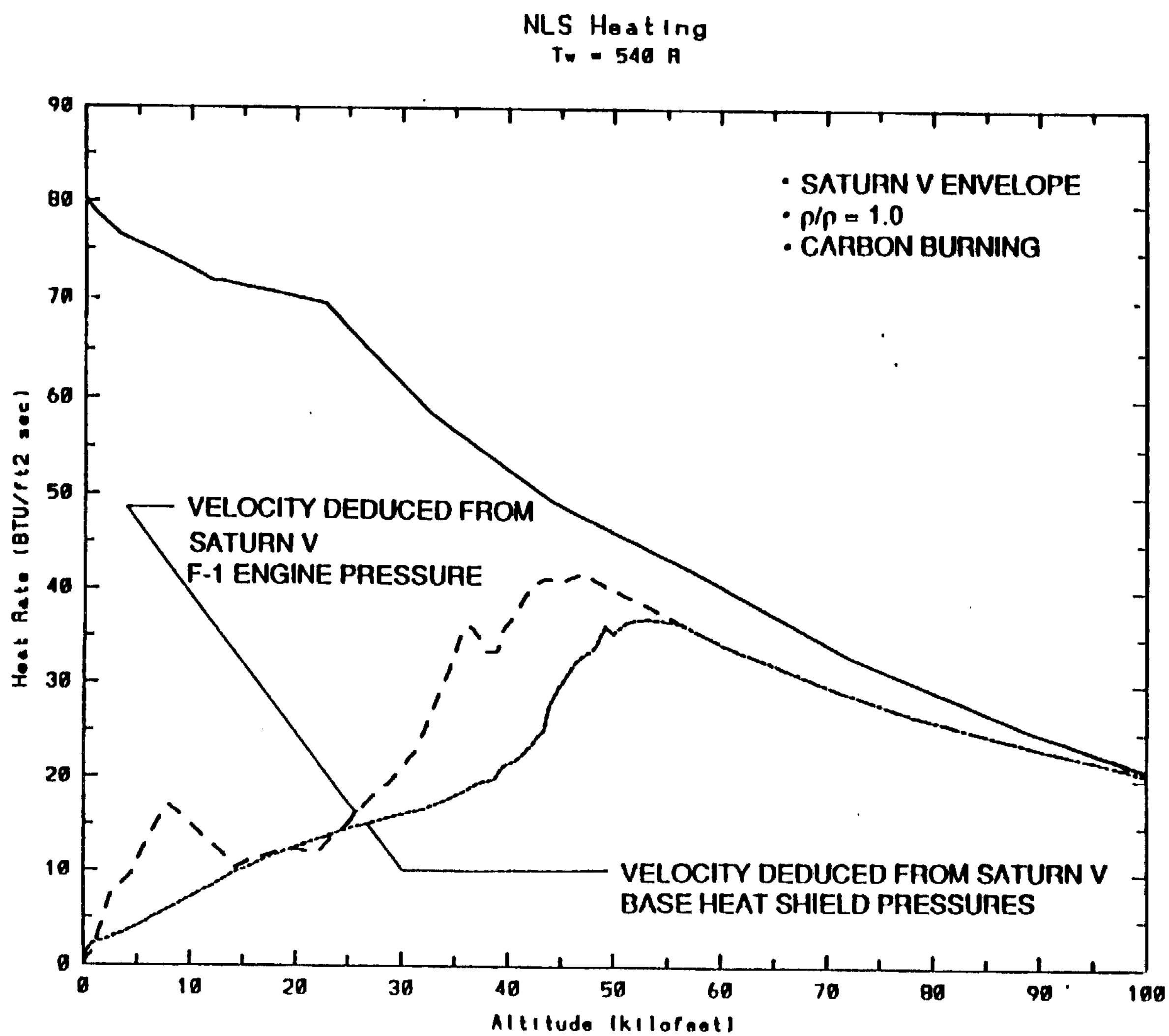
BASE HEAT SHIELD



EFFECT OF TRAJECTORY ON BASE REGION CONVECTIVE HEAT TRANSFER COEFFICIENT

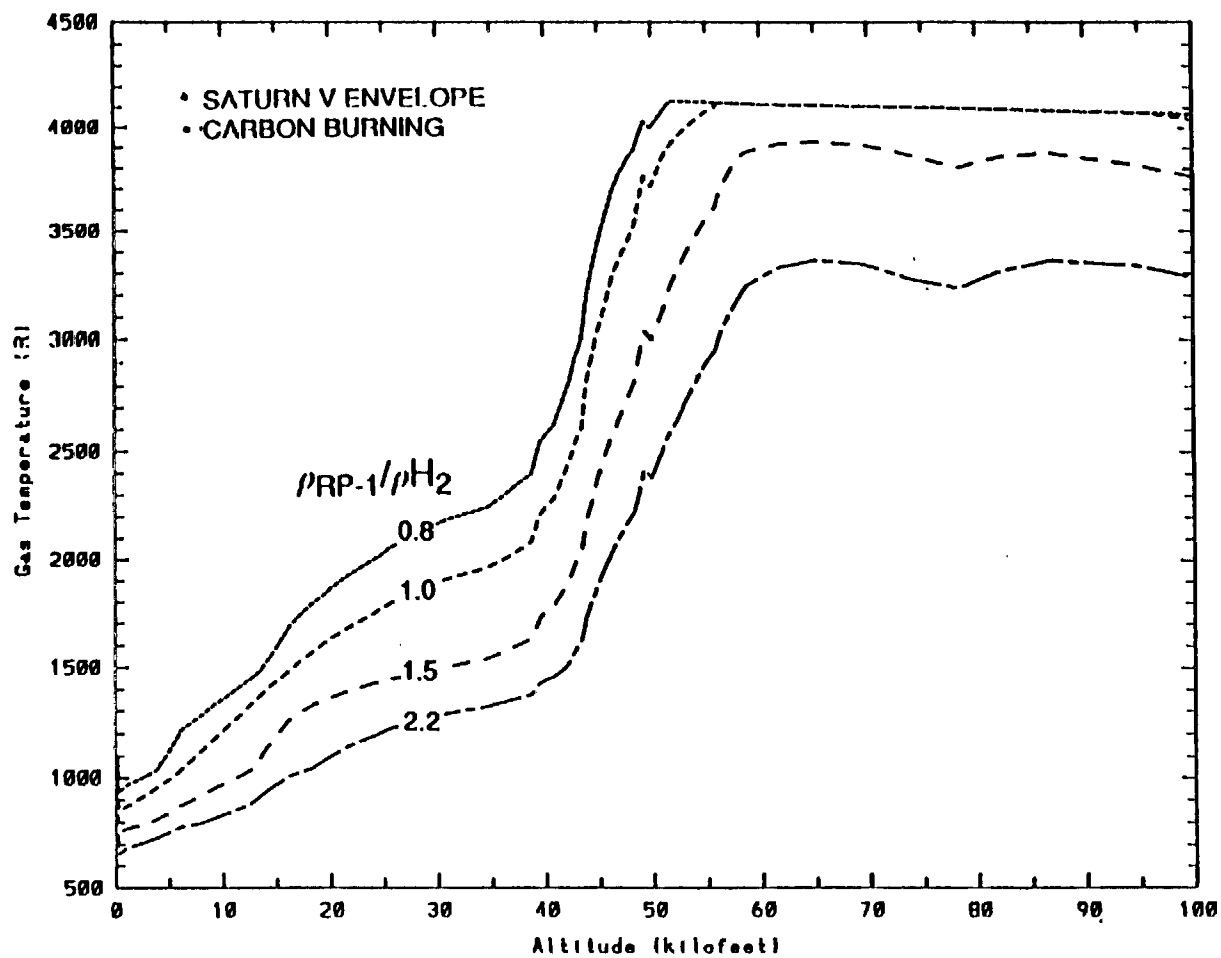


EFFECT OF BASE GAS VELOCITY ON h_c WITH REYNOLDS NUMBER ADJUSTMENT

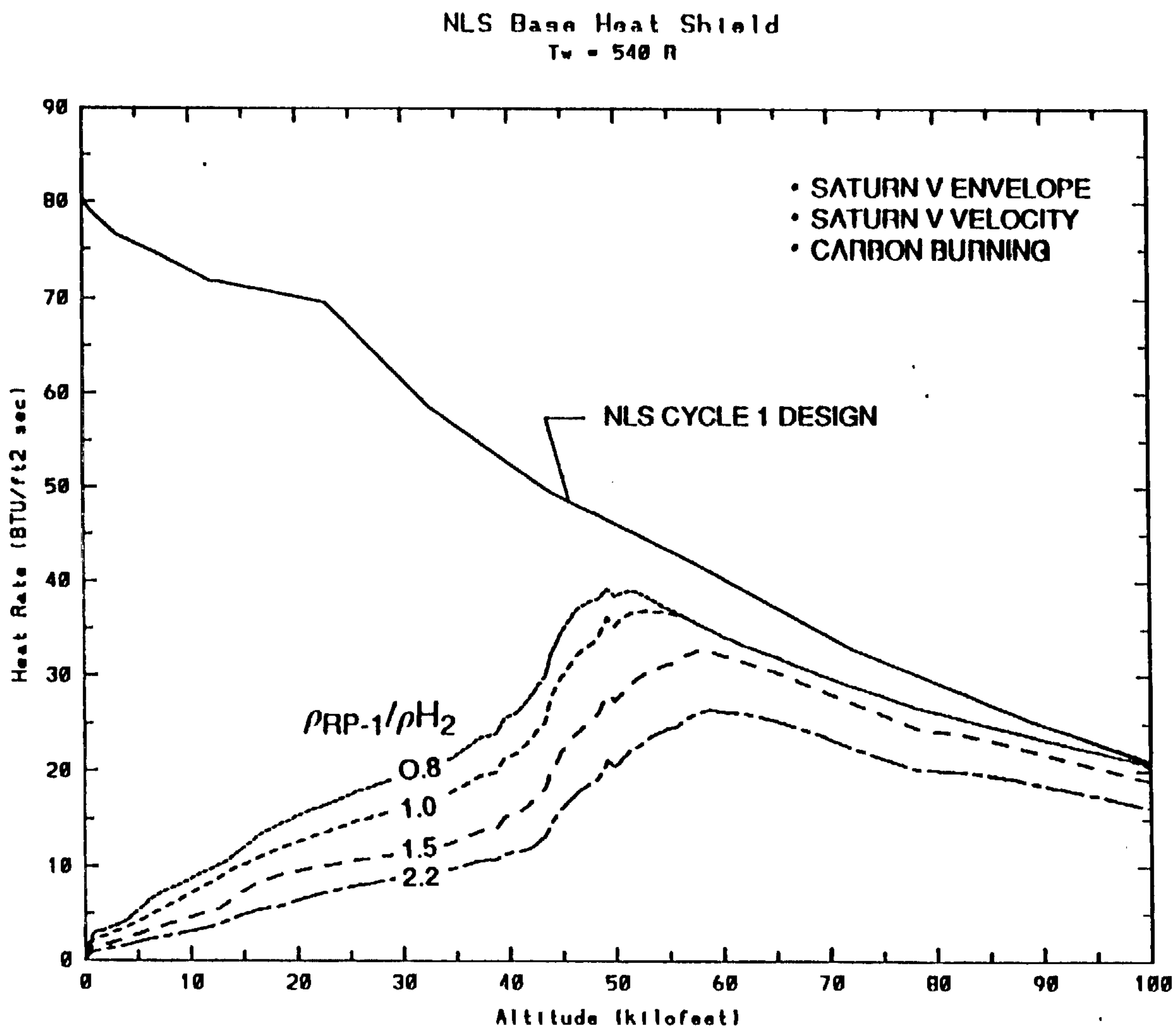


SCALING SATURN GAS TEMPERATURES TO NLS
VARIATION IN ENTRAINMENT DENSITY RATIO CORRECTION

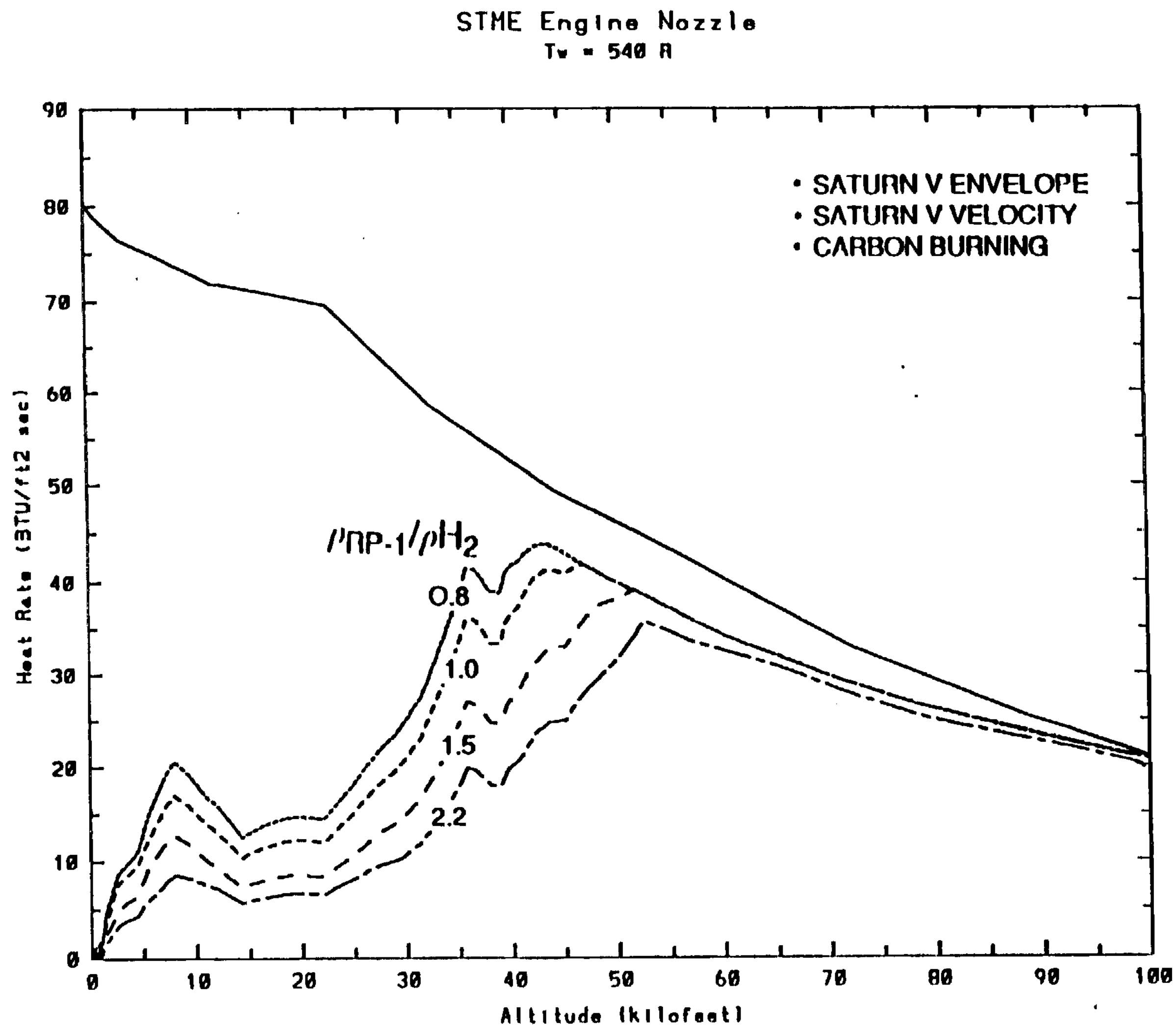
NLS Base Heat Shield



SCALING SATURN GAS TEMPERATURES TO NLS
VARIATION IN ENTRAINMENT DENSITY RATIO CORRECTION



SCALING SATURN GAS TEMPERATURES TO NLS
VARIATION IN ENTRAINMENT DENSITY RATIO CORRECTION



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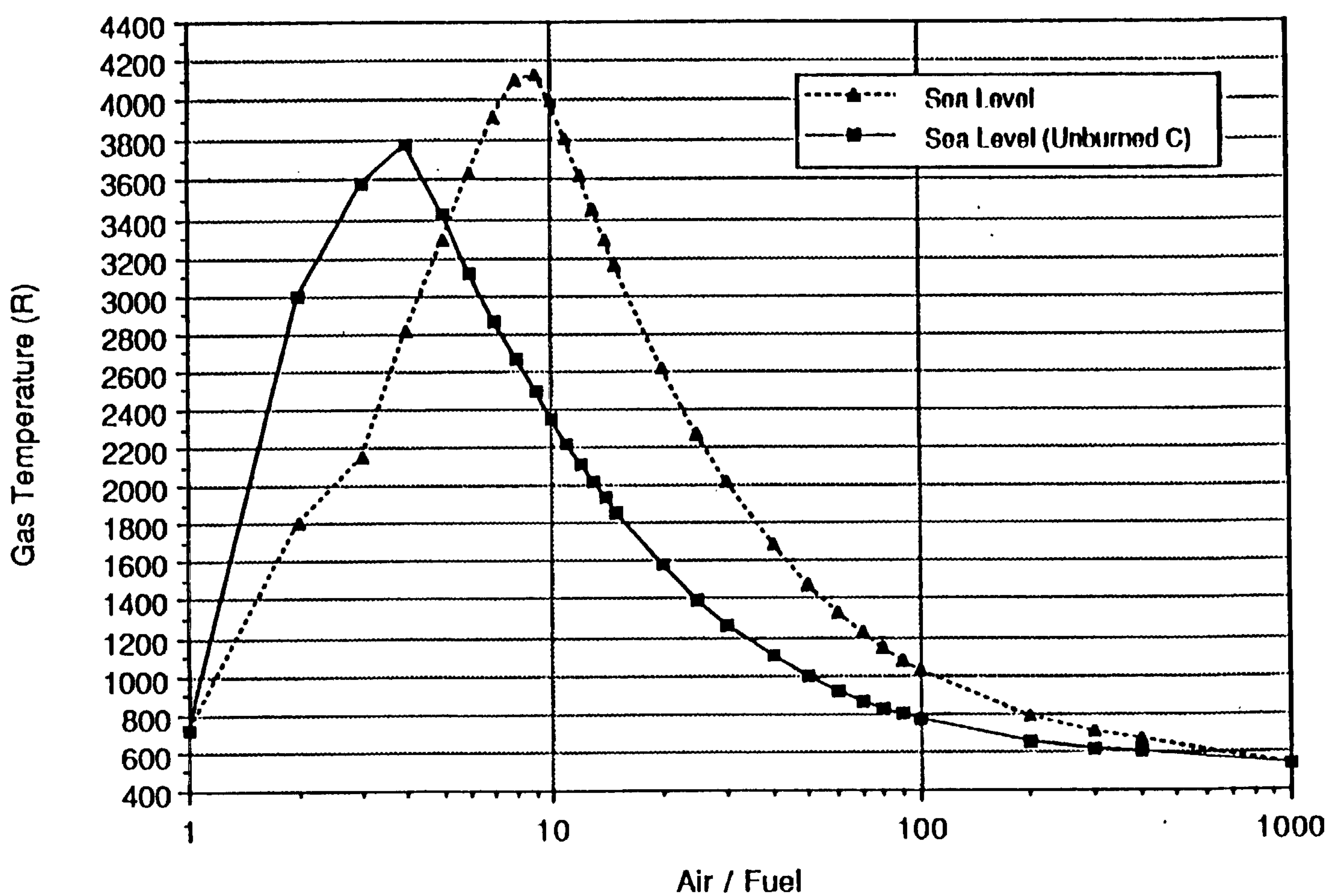
F-1 ENGINE TURBINE EXHAUST COMBUSTION TEMPERATURES WITH AIR

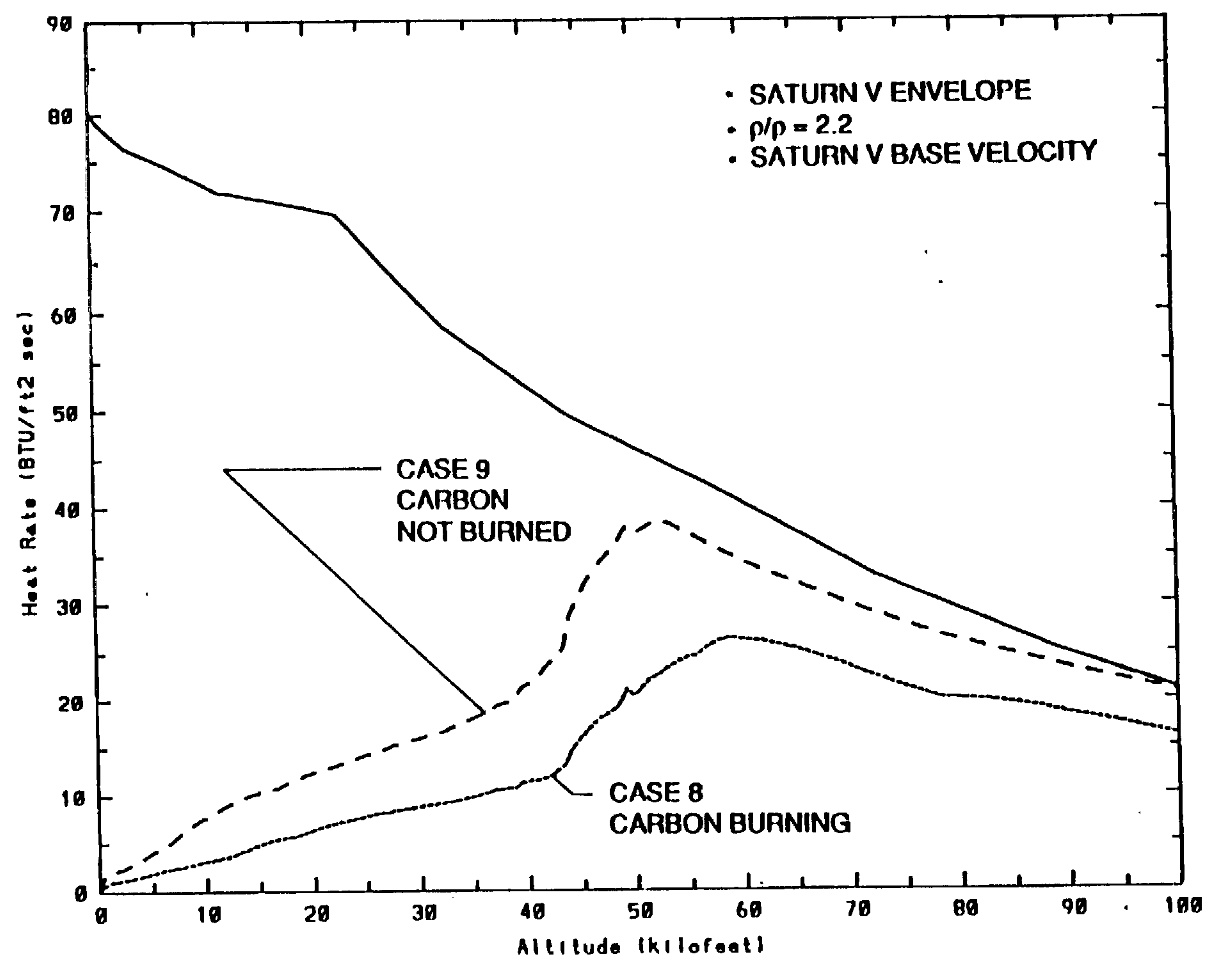
CEC OUTPUT

Fuel: F1 Turbine Exhaust (Tinit=791 K)
Oxidizer: Air (Tinit at Ambient Conditions)



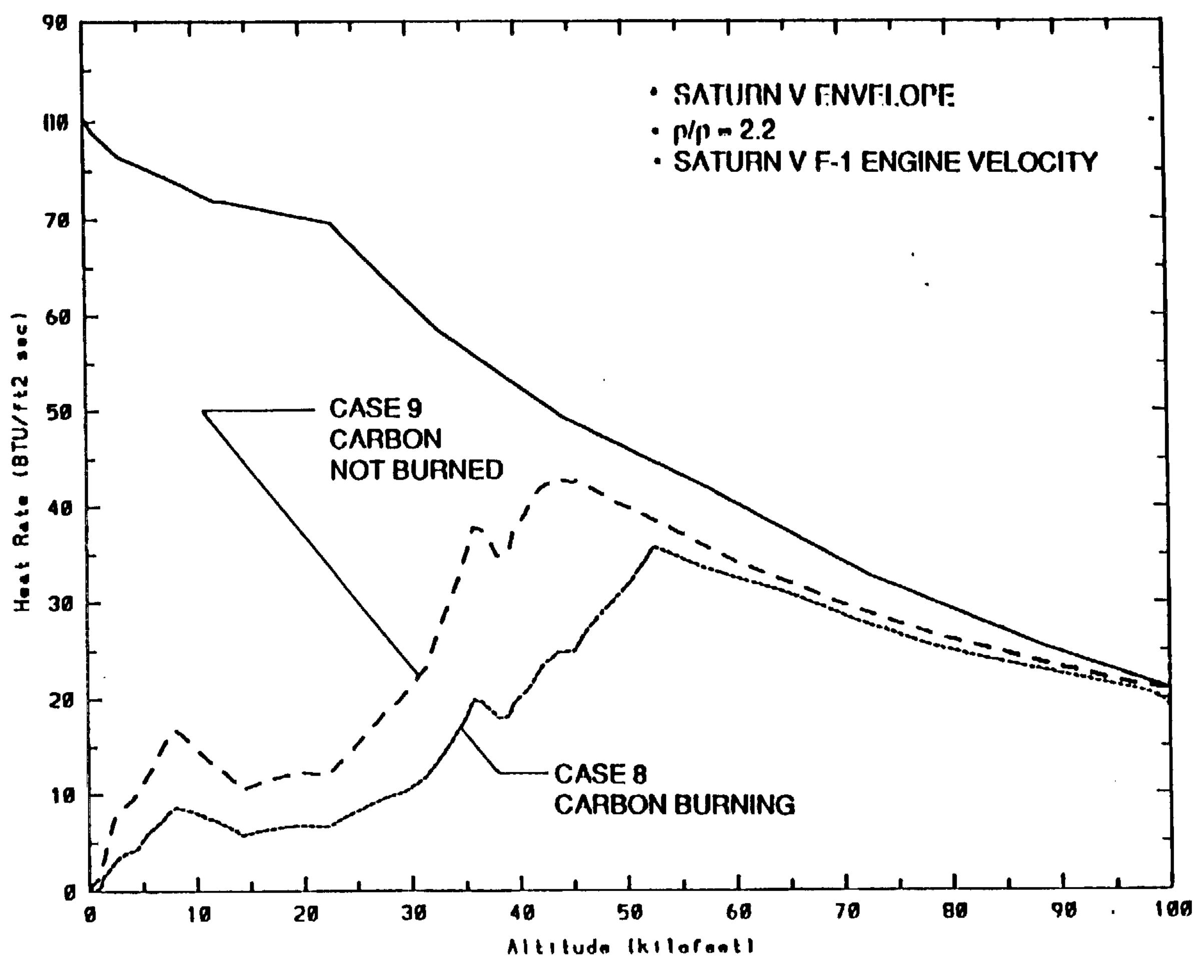
EFFECT OF CARBON BURNING ASSUMPTION ON NLS CONVECTIVE HEATING RATES

NLS Base Heat Shield
Tw = 540 R



EFFECT OF CARBON BURNING ASSUMPTION ON NLS CONVECTIVE HEATING RATES

STME Engine Nozzle
Tw = 540 R



COMPARISON OF CONVECTIVE HEATING ENVIRONMENTS

1.5 STAGE BASE HEAT SHIELD

CASE #	METHODOLOGY										ENVIRONMENT		RATIO TO NOMINAL	
	ENVELOPE		V BASE		ρ/ρ				CARBON		PEAK	LOAD[2]	PEAK	LOAD
	SAT V	ALLSAT	$F(V_\infty)$	SAT V	0.8	1.0	1.5	2.2	Burn	Unburn	qc[1]			
Reference	x			x		x			x		36.9	1721	1.00	1.00
1		x		x		x			x		36.9	2258	1.00	1.31
2	x		x			x			x		36.9	2002	1.00	1.16
3	x			x	x				x		39.2	1884	1.06	1.09
4	x			x			x		x		32.7	1416	0.89	0.82
5	x			x				x	x		26.5	1107	0.72	0.64
6	x			x		x				x	43.5	2342	1.18	1.36
7		x	x		x					x	72.9	4709	1.98	2.74
8	x			x				x	x		26.5	1107	0.72	0.64
9	x			x				x		x	38.3	1733	1.04	1.01

[1] BTU/ft^2-sec
[2] BTU/ft^2

Carbon Unburned
Most Realistic (Design)

COMPARISON OF CONVECTIVE HEATING ENVIRONMENTS

STME NOZZLE

	METHODOLOGY										ENVIRONMENT		RATIO TO NOMINAL	
CASE #	ENVELOPE		V BASE		ρ/ρ				CARBON		PEAK qc[1]	LOAD[2]	PEAK	LOAD
	SAT V	ALLSAT	F(V∞)	SAT V	0.8	1.0	1.5	2.2	Burn	Unburn				
Reference	x			x		x			x		41.7	2106	1.00	1.00
1		x		x		x			x		41.7	2389	1.00	1.13
2	x		x			x			x		41.7	2586	1.00	1.23
3	x			x	x				x		43.9	2280	1.05	1.08
4	x			x			x		x		39.0	1813	0.94	0.86
5	x			x				x	x		35.7	1533	0.86	0.73
6	x			x		x				x	47.1	2749	1.13	1.31
7		x	x		x					x	72.9	4713	1.75	2.24
8	x			x				x	x		35.7	1533	0.86	0.73
9	x			x				x		x	42.8	2126	1.03	1.01

[1] BTU/ft^2-sec
[2] BTU/ft^2

Carbon Unburned
Most Realistic (Design)

REMTECH

NLS BASE HEATING
TECHNICAL INTERCHANGE MEETING

Presented by:
ROBERT L. BENDER
REMTECH Inc.

JULY 16, 1992

2

PRESENTATION OUTLINE

- Background/Problem Definition
- Saturn Flight Data Review
- NLS 1.5 Stage Convective Base Heating Methodology
- NLS 1.5 Stage Convective Base Heating Environments
- Conclusions

BACKGROUND/PROBLEM DEFINITION

BASE HEATING ENVIRONMENT COMPONENTS

The base heating environment is composed of a convective heating component and radiation component. Convection occurs as the base region gases flow over the base structure. Radiation to the base may be the combined radiation from several sources including: the core of the down-stream plumes, the plume mixing boundaries, plume interaction regions, local hot gases in the base, localized burning in the base, or, occasionally, from other hot structures in the base. Most analysts are concerned with main plume radiation and convective heating from reversed gases.

RADIATION SOURCES:

- *Low Altitude* (< 70 kft)
  - Plume Core (Mach Disk)
  - Afterburning
  - Baseburning (Turbine Exhaust)
- *High Altitude (> 70 kft)*
  - Plume Core (Near Field)
  - Plume Interaction Zones
  - Base Recirculation
- *SRM Shutdown Spike*

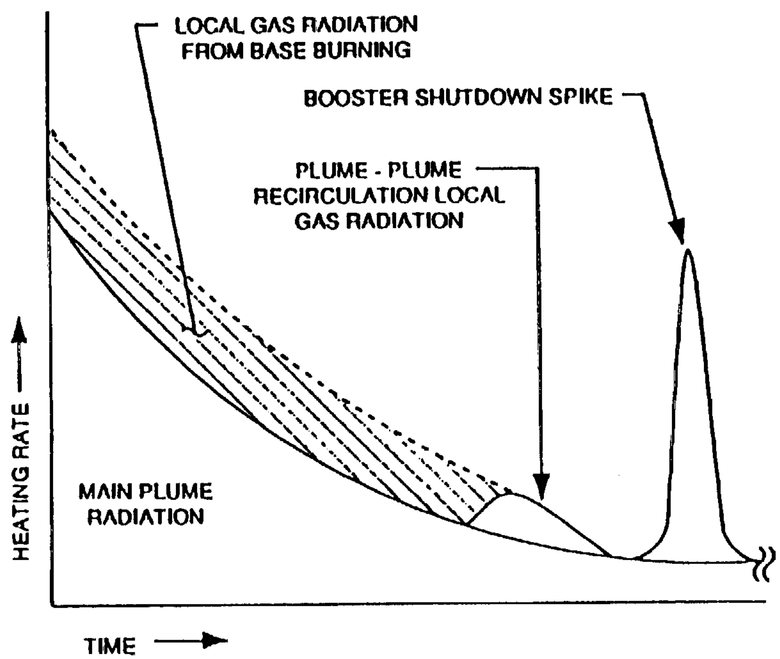
CONVECTION SOURCES:

- *Cooling from Ambient Air*
- *Heating from Recirculated Plume Gases*
  - Plume - Plume Interactions
  - Plume - Freestream Interactions
- *Base Burning from Recirculated Turbine Exhaust*

BASE BURNING vs CONVENTIONAL BASE HEATING

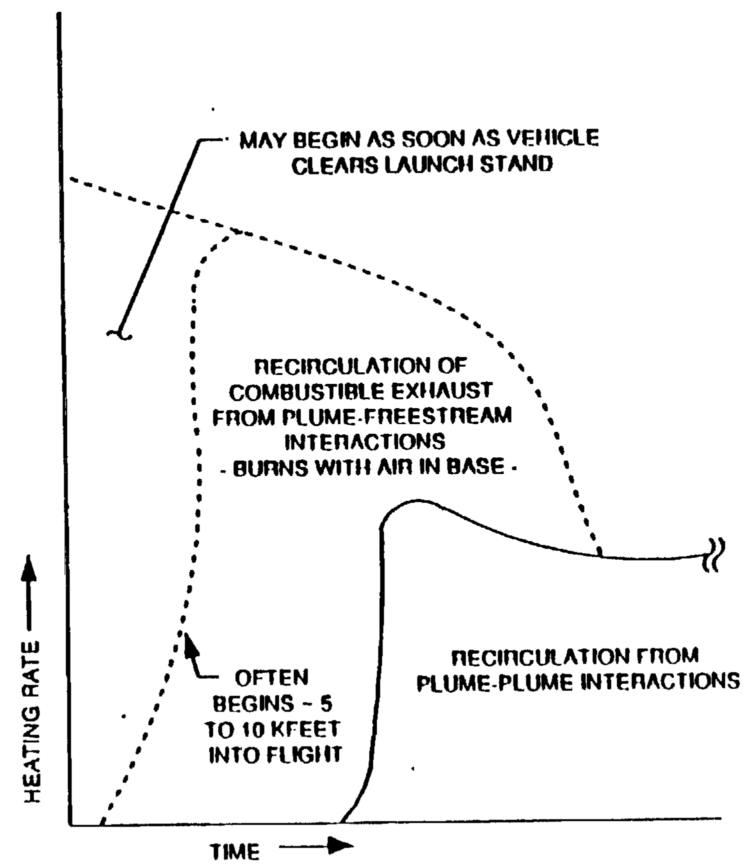
RADIATION

- Base burning increase in radiation normally small compared with conventional radiation



CONVECTION

- Base burning convection may be large in relation to conventional convection



PAST EXPERIENCE WITH TURBINE EXHAUST DISPOSAL
--- LARGE U.S. LAUNCH VEHICLES ---

VEHICLE	T.E. DISPOSAL SCHEME	EXPERIENCE/LESSON LEARNED
JUPITER -1A	• Duct Along Nozzle to Exit Plane • Change to Outboard Duct	• 1st Flight Failed Due to Base Heating • No failure
ATLAS	• Duct into Base - By Center Engine • Change to Outboard Duct	• 1st 2 Flights Failed Due to Base Heating • No Failure
DELTA	• Duct through Heat Shield	• High local heating on heat shield while SRM's attached
TITAN II	• Two ducts exiting slightly aft of boattail base. • Strong air scooping eliminates base burning.	• Heating not severe • No failure due to T.E. burning
TITAN III (Core)	• Core engine ignited at H $\geq$ 100 kft; above altitude of serious burning.	• No trouble
SATURN I	• Inbd engine ducted to fin outbd of base • Outbd engine into nozzle through exhausterator.	• High heating early in flight • No failure due to T.E. burning
SATURN IB	• Inbd engine ducted through 4 crescent opening in flame shield • Exhausterator on outbd engine	• T.E. exhaust did not burn; cooled flame shield • No failure
SATURN V	S-IC Stage — F-1 Engine T.E. Dumped in Nozzle @ $A/A^*=10$	• No Failure Due to Base Heating • Unburned RP-1 Afterburning in Plume @ Low Altitude, Burned in Base @ High Altitude
NSTS SPACE SHUTTLE	• No T.E. Disposal on SSME • SRB T.E. Dumped Outboard	• No Failure Due to Base Heating • Predictable Environments

SUMMARY OF TURBINE EXHAUST DISPOSAL
- LARGE U.S. LAUNCH VEHICLES -

- Flight vehicles with turbine exhaust disposal into base, engine nozzle, or external flow.
  - ATLAS
  - SATURN 1 & 1B, 1st Stage
  - SATURN V, 1st Stage
  - DELTA

  } LO_2/RP-1 Propellants

  - TITAN — Aerozine 50/UDMH Propellants (Storable)
- Flight vehicles which utilized LO_2/LH_2 propellants.
  - S–IV Stage, SATURN 1
  - S–II Stage, SATURN V
  - S-IV B Stage, SATURN V

  } T.E. Dumped inside nozzle-high altitude.

  - Shuttle Orbiter — Regeneratively cooled nozzle — no T.E. Discharge

THE NLS-STME TURBINE EXHAUST
· BASE BURNING POTENTIAL

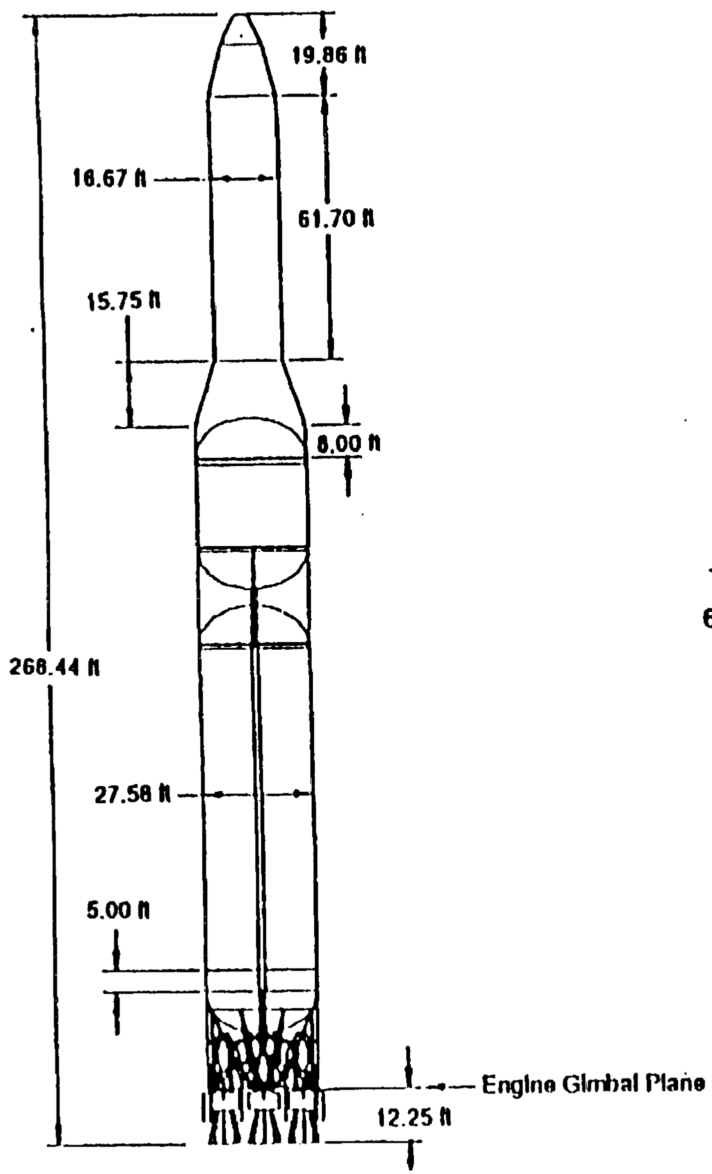
- ***The STME with film/convective dump cooled nozzle:***

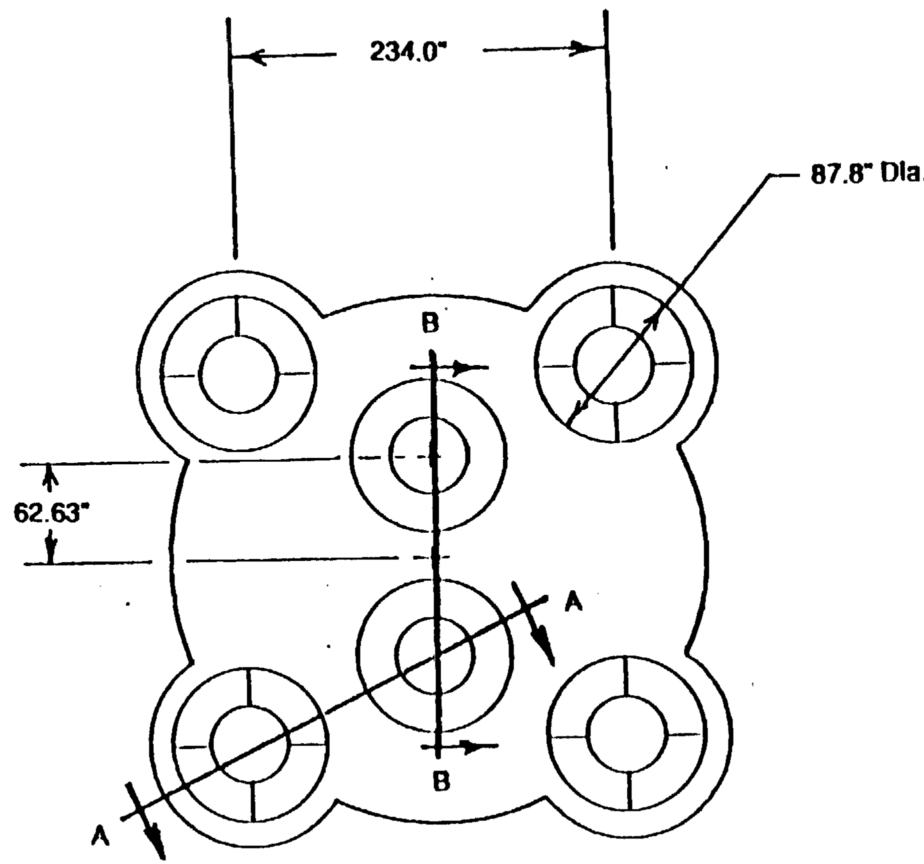
  - is a new concept, outside experience range
  - creates potential for large mass flow of energy, unburned H_2 at nozzle exit lip
  - H_2 will burn over wide range of mixture ratios (and pressures) with oxygen (air) present in base

- ***The 1.5 stage:***

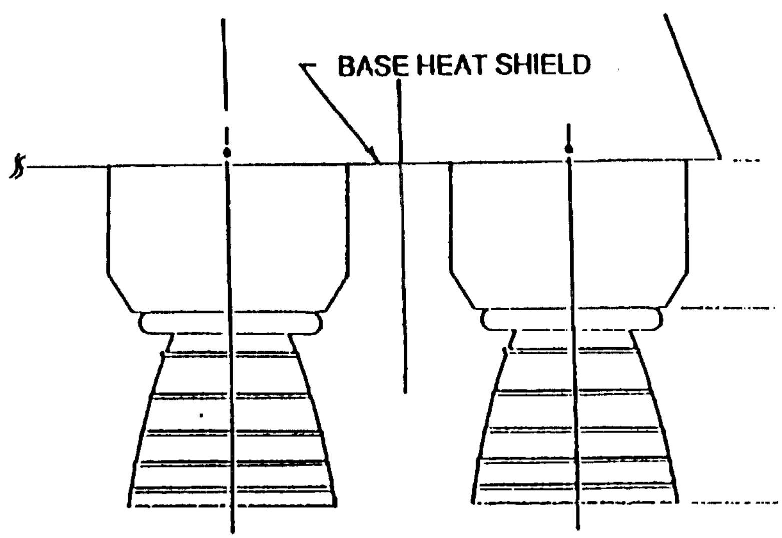
  - will have complex base flowfields and low altitude recirculation of plume boundary gases near the nozzle exit

1.5 STAGE REFERENCE VEHICLE
BASE GEOMETRY

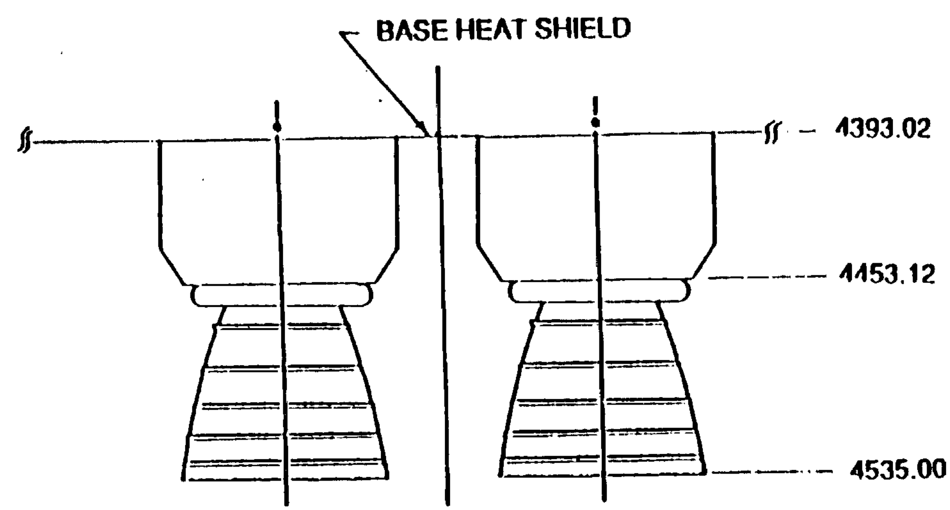




1.5 STAGE REFERENCE VEHICLE
BASE REGION SIDE VIEWS

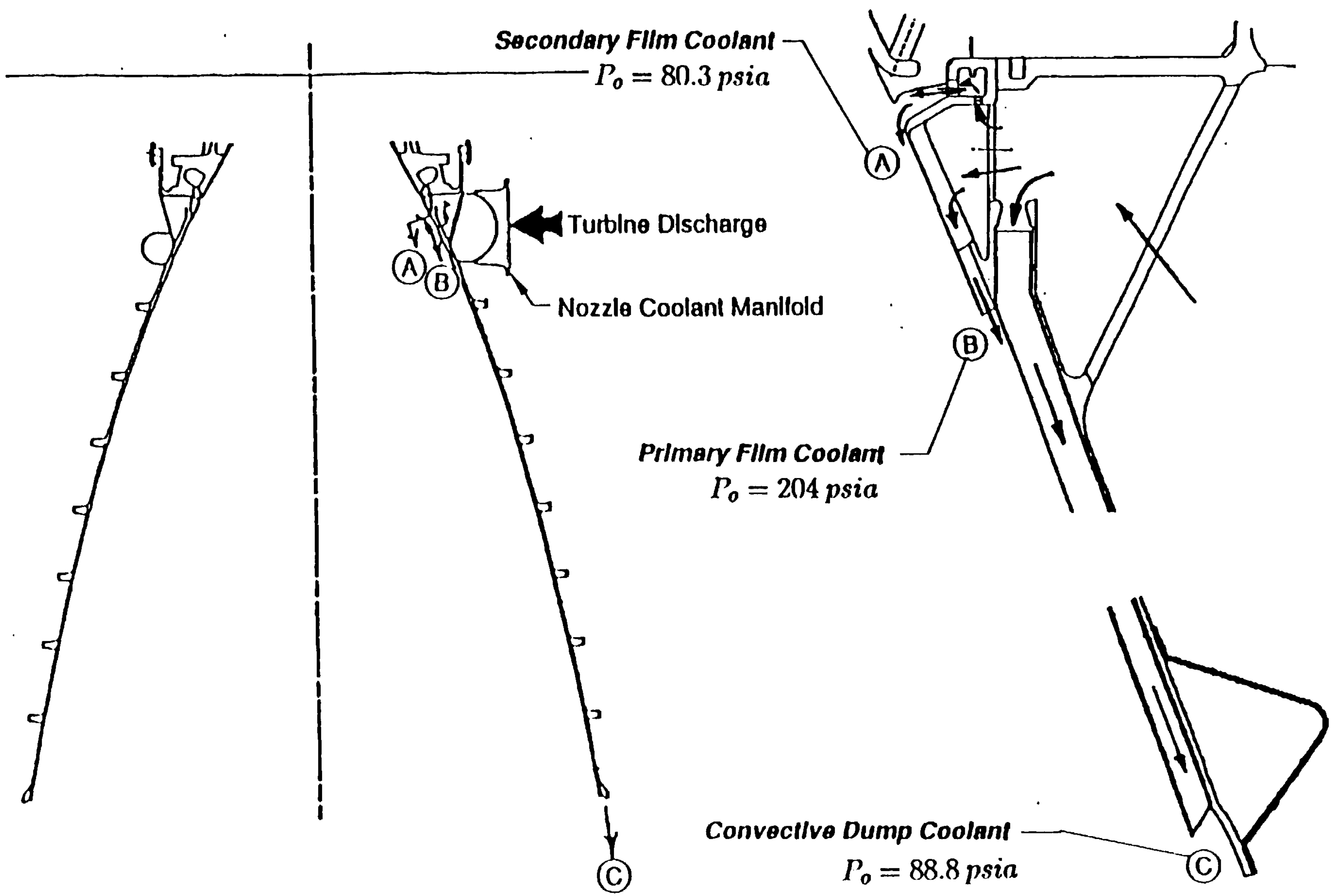


1.5 STAGE
VIEW A - A

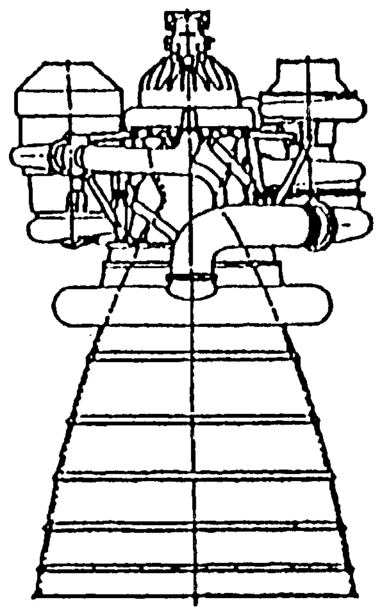


1.5 STAGE
VIEW B - B

STME TURBINE EXHAUST DISCHARGE



STME FILM/CONVECTIVE DUMP COOLED NOZZLE TURBINE EXHAUST FLOW RATES



MAIN CHAMBER

$P_o - 2250^o\ psia$

$T_o - 6708^o\ R$

$\dot{\omega} - 1292.7\ lbm/sec$

TURBINE EXHAUST DISCHARGE

(B) •Primary Film Coolant.

$P_o - 204\ psia$

$T_o - 1190^o\ R$

$\dot{\omega} - 24.4\ lbm/sec$

(A) •Secondary Film Coolant

$P_o - 80.3\ psia$

$T_o - 1190\ R$

$\dot{\omega} - 4.26\ lbm/sec$

(C) •Convective Coolant

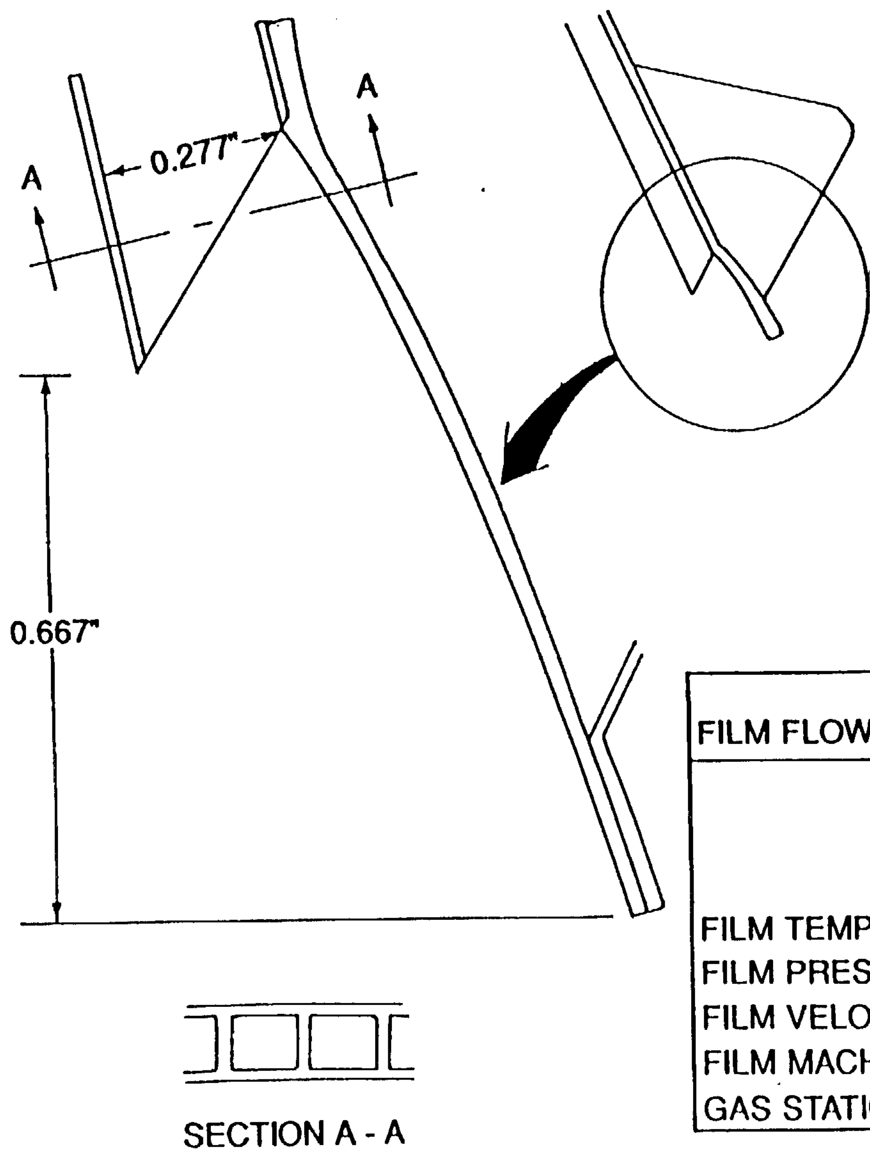
$P_o - 88.8\ psia$

$T_o - 1462.4^o\ R$

$\dot{\omega} - 35.4\ lbm/sec$

NOTE: Turbine exhaust Is: 47% H_2
53% H_2O (Steam)
mass percent

STME TURBINE EXHAUST DISCHARGE AT NOZZLE EXIT LIP



DUMP INJECTOR GEOMETRY

NUMBER OF SLOTS:	570
SLOT HEIGHT @ throat (IN):	0.2770
SLOT WIDTH @ throat (IN):	0.4570
WEB WIDTH (IN):	0.0600
LIP THICKNESS (IN):	0.0300
STEP HEIGHT (IN):	0.3070
SLOT THROAT AREA (IN2):	72.1557
SLOT EXIT AREA (IN2):	133.4874

DUMP INJECTOR PERFORMANCE

FILM FLOW RATE (lbm/sec)	26.527		
	INJECTOR STATIONS		
	INLET	THROAT	EXIT
FILM TEMPERATURE (R):	1469.0	1237.0	806.7
FILM PRESSURE (PSIA):	73.7	39.2	8.2
FILM VELOCITY (ft/sec):		4726.8	7986.3
FILM MACH NUMBER:		1.000	2.092
GAS STATIC PRESSURE:			8.14

CHRONOLOGY OF NLS ENVIRONMENT PREDICTIONS LEADING TO CYCLE 1

PRELIMINARY CYCLE 1: ED33 (98 - 91) September 1991

- Base burning effect on convection not considered
- Base burning effect on radiation approximated

PRELIMINARY CYCLE 1 UPDATE: ED33 (03 - 92) JANUARY 1992

- Base burning effect on convection from ED31 (06 - 89) gas temperature and Saturn I heat transfer coefficient
- base burning effect on radiation same as ED33 (98 - 91)

CYCLE 1 ENVIRONMENT ED33 (15 - 92) February 1992

- Base burning effect on convection updated to stoichiometric air - T.E. exhaust mixture at estimated base pressure; Saturn I heat transfer coefficient retained. (Reduction in Δ enthalpy from ED33 [03 - 92])
- Local radiation in base estimated from base burning gas composition and thermodynamic properties
- Main plumes radiation from updated plume models

PROBLEM DEFINITION

- Cycle 1 quick look/upper limit base convective heating environment:
  - assumed H_2 burned stoichiometrically in base resulting in temperatures approaching 4200°R.
  - assumed film heat transfer coefficient based on Saturn I averaged results defined in Saturn base heating handbook.
  - did not have experimental data or CFD results to validate levels of temperature or heating.
- These Cycle 1 environments resulted in significant thermal design penalties for the base region including the STME.

NLS TURBINE EXHAUST BASE BURNING ANALYSIS PLAN FOLLOWING PUBLICATION OF CYCLE 1 ENVIRONMENTS

- Continue to analyze and refine plume definitions and base flowfield thermochemistry data.
- Continue analysis of previous launch vehicle experience with various turbine exhaust disposal schemes with emphasis on Saturn flight vehicles.
- Coordinate base heating studies with STME design evolution.
- Outline test program to provide explicit thermal environment data for NLS configurations, trajectories, and STME turbine exhaust disposal schemes.
- Provide up-dated base heating environments as needed to support design evolution.

CHRONOLOGY OF ACTIVITIES FOLLOWING CYCLE 1

ACTIVITY	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep
	CY 1992							
• *CYCLE 1 DESIGN ENVIRONMENT*								
o ED 33 (15-92) Memo Out	2/2							
o NLS Chief Engineer Briefing	2/8							
o NLS Contractor Briefing	2/20							
• *FOLLOW-ON ANALYSIS*								
o Saturn Data Review								
- Begin		3/2						
- Task 1-h_c From Flight								
- Task 2 - T_{GAS} From Saturn V								
- ED31 Briefing				5/19				
o Methodology Development								
- Begin				5/20				
- Saturn V Scaling								
- Working Group Briefing					6/24			
• *NLS 1.5 STAGE ENVIRONMENTS*								
o Environment Definition								
o ED01 Lab Briefing						7/01		
o MSFC S&E Briefing						7/08		
o Publish Environments						7/31		

C-4

REMTECH

SATURN FLIGHT DATA REVIEW

19.

SATURN FLIGHT DATA REVIEW AND ANALYSIS OBJECTIVES

A detailed review of applicable Saturn flight data during early ascent was recommended to verify or improve the Cycle 1 environment.

OBJECTIVE 1: Reduce heat load accumulated early in flight by refining Cycle 1 film heat transfer coefficient (h_2) based on Saturn flight data

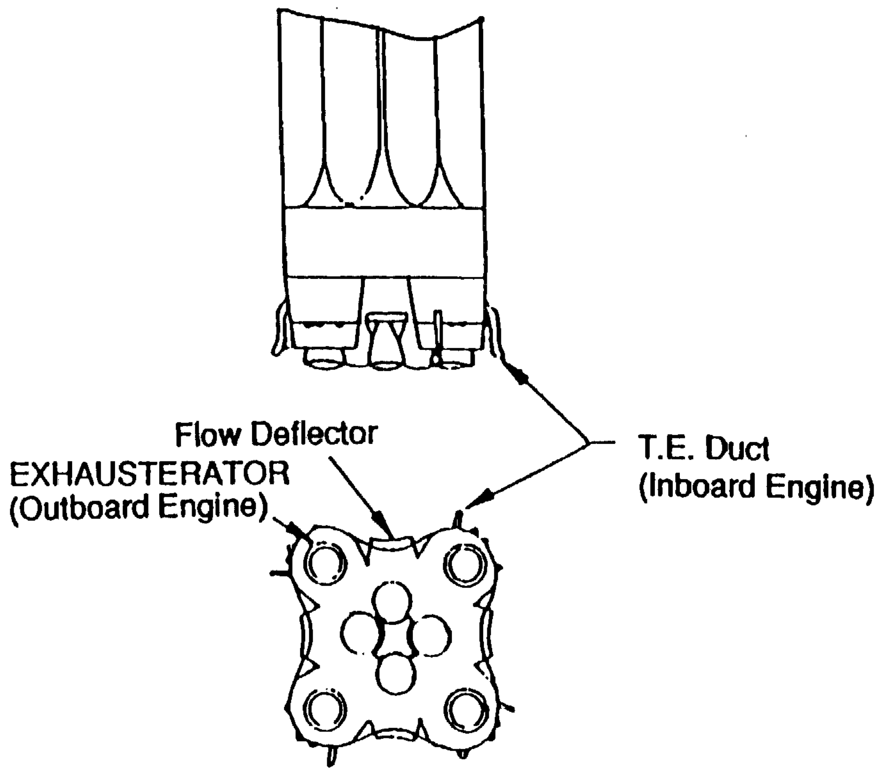
$$h_c = \frac{Q_{Total} - Q_{Radiation}}{T_{Gas} - T_{Wall\ of\ Total\ Cal}}$$

OBJECTIVE 2: Based on Saturn flight data, develop a methodology to reduce base gas temperature early in flight below stoichiometric if possible

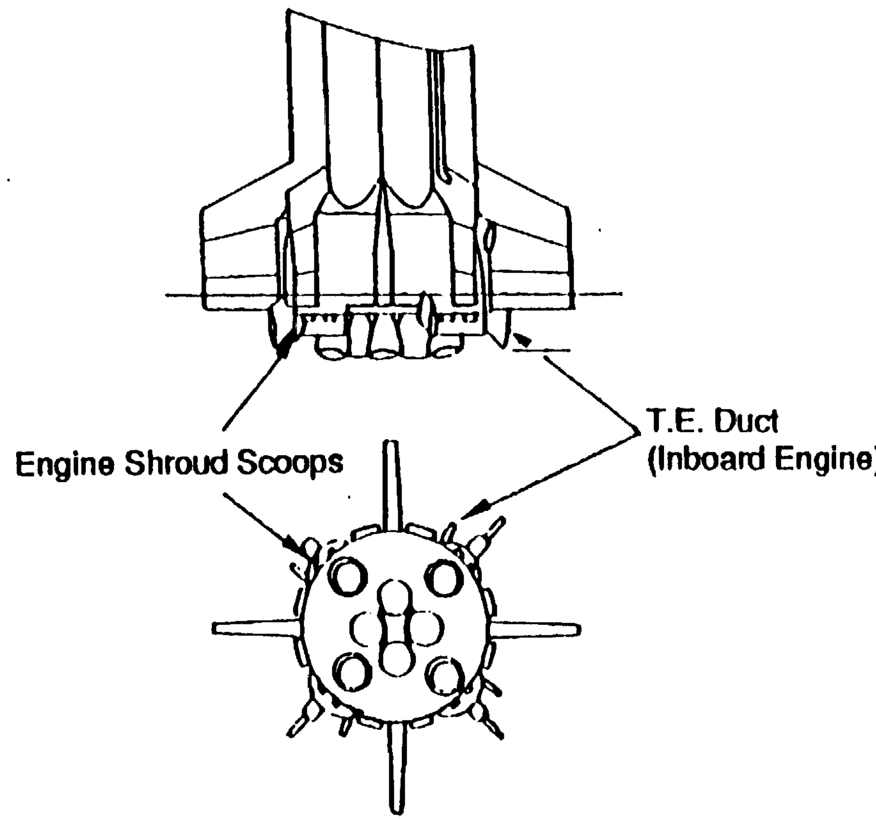
SUMMARY OF SATURN FLIGHT VEHICLES

VEHICLE	FIRST STAGE DESIGNATION	FLIGHT DESIGNATION	FIRST STAGE ENGINE DESCRIPTION				
			DESIG	PROP	THRUST	NO.	TYPE
Saturn I Block I	S-I	4 Flights SA-I to SA-4	H-1	LOX/RP-1	165K	8	Gas Generator Cycle
Saturn I Block II	S-I	6 Flights SA-5 to SA-10	H-1	LOX/RP-1	188K	8	Gas Generator Cycle
Saturn IB	S-IB	4 Flights AS-201 to AS-204	H-1	LOX/RP-1	200K	8	Gas Generator Cycle
Saturn V	S-IC	12 Flights AS-501 to AS-512	F-1	LOX/RP-1	1.53M	5	Gas Generator Cycle

SATURN FLIGHT VEHICLE BASE CONFIGURATIONS

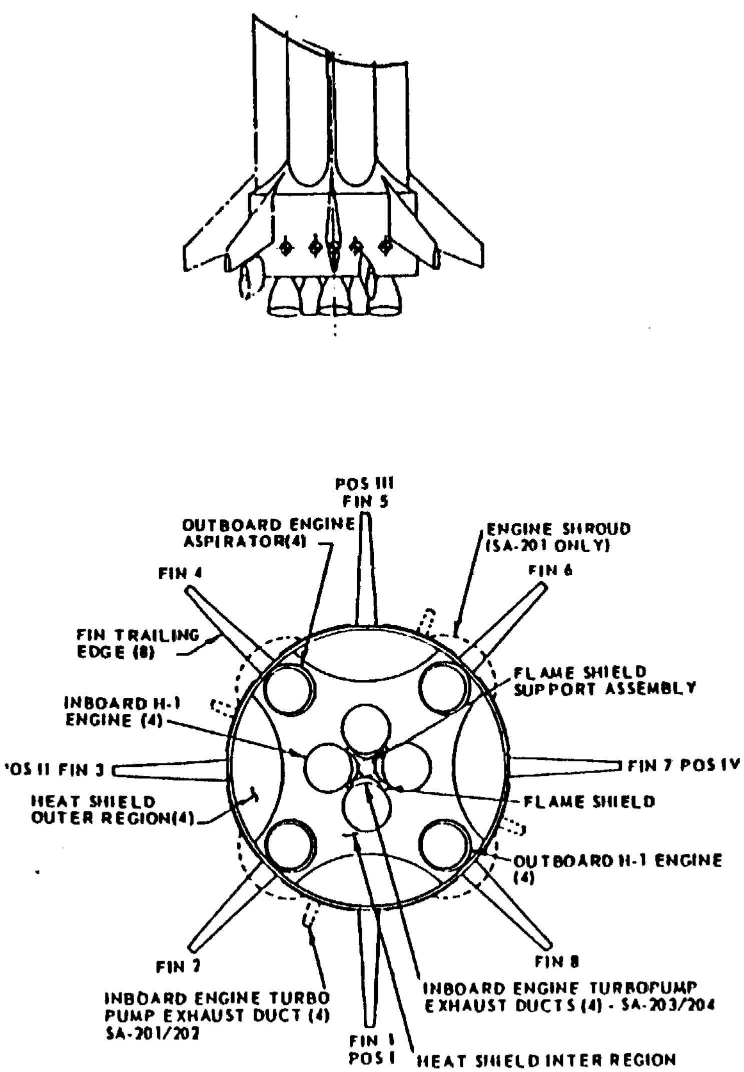


SATURN I, BLOCK I

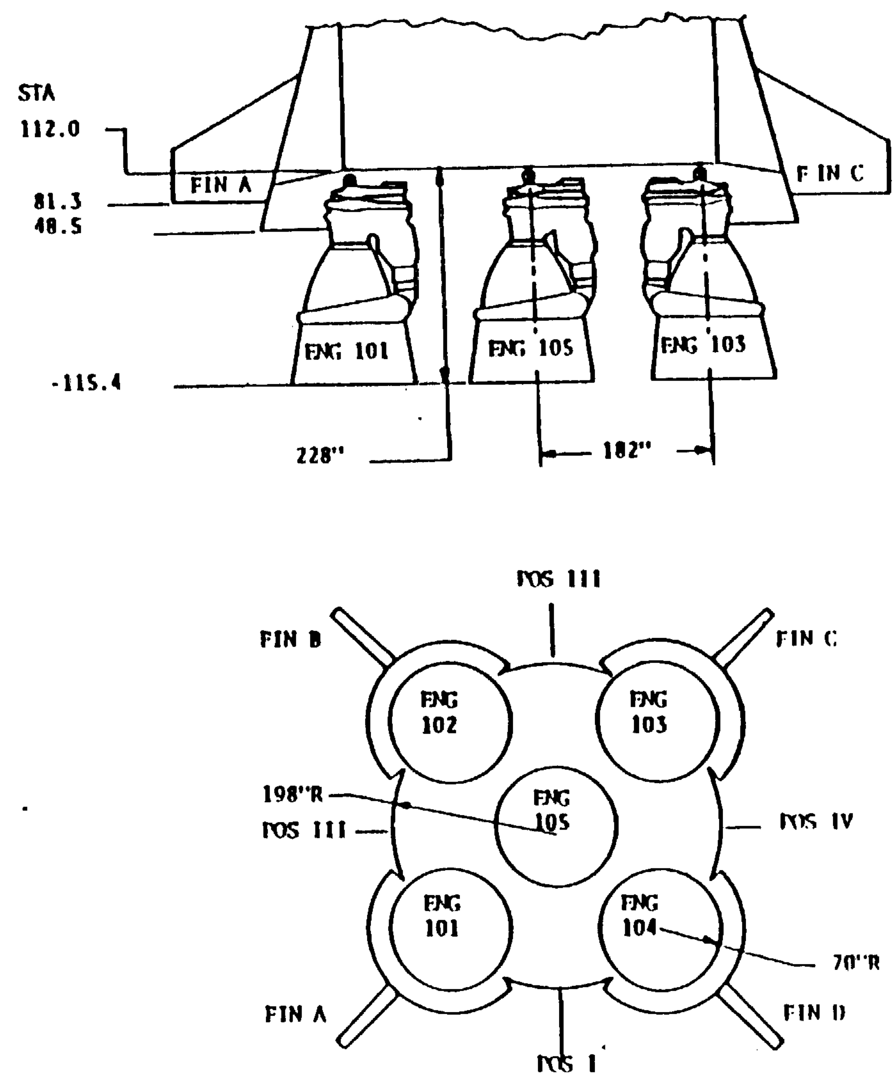


SATURN I, BLOCK II

SATURN FLIGHT VEHICLE BASE CONFIGURATIONS

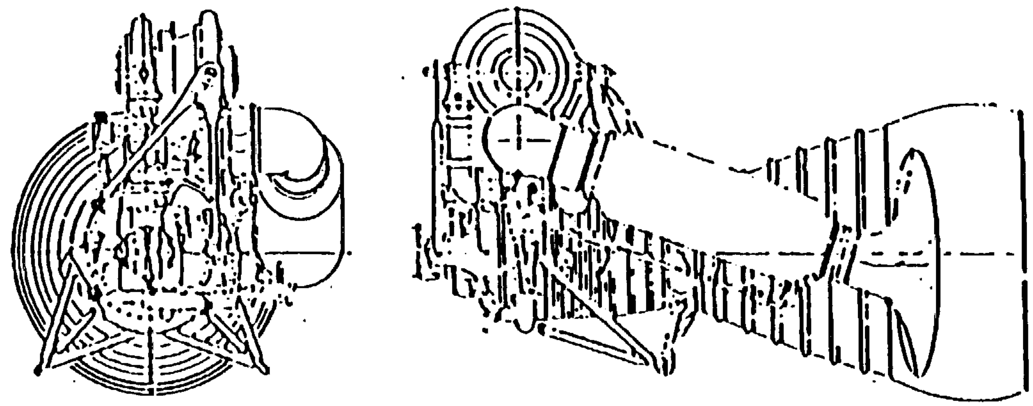


SATURN IB

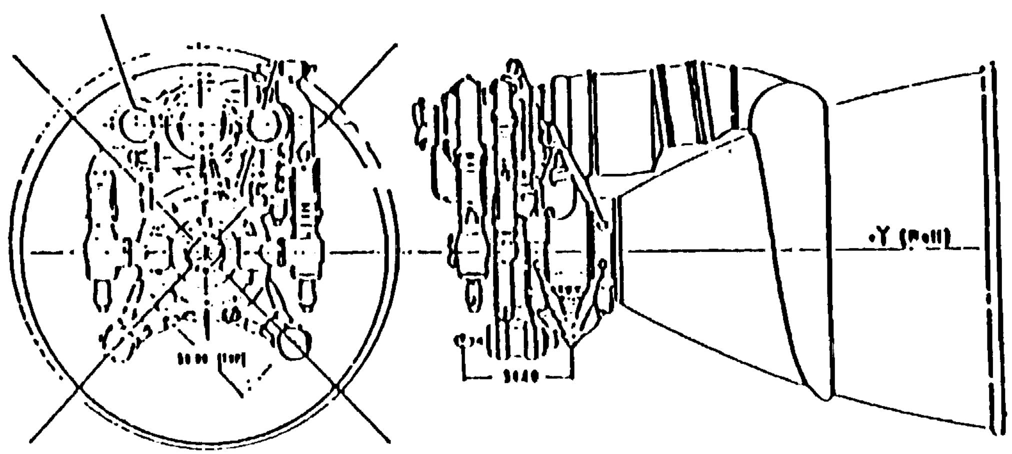


SATURN V

TURBINE EXHAUST DISPOSAL INSIDE NOZZLE

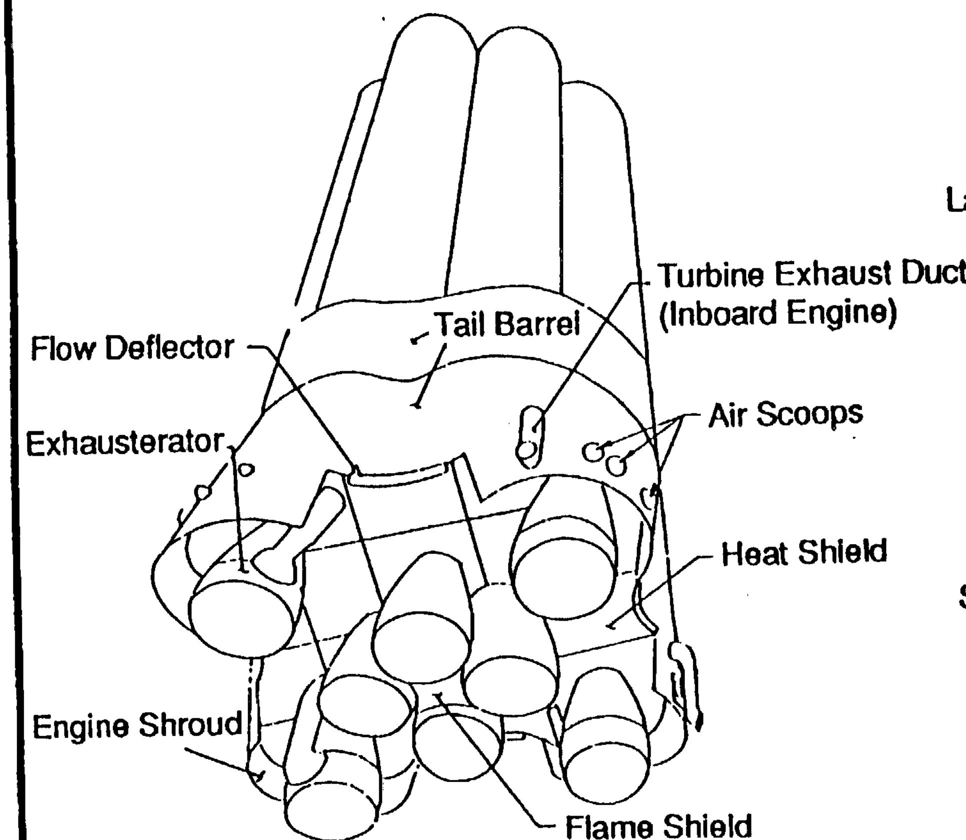


SATURN I AND IB BOOSTERS - OUTBOARD H-1 ENGINE

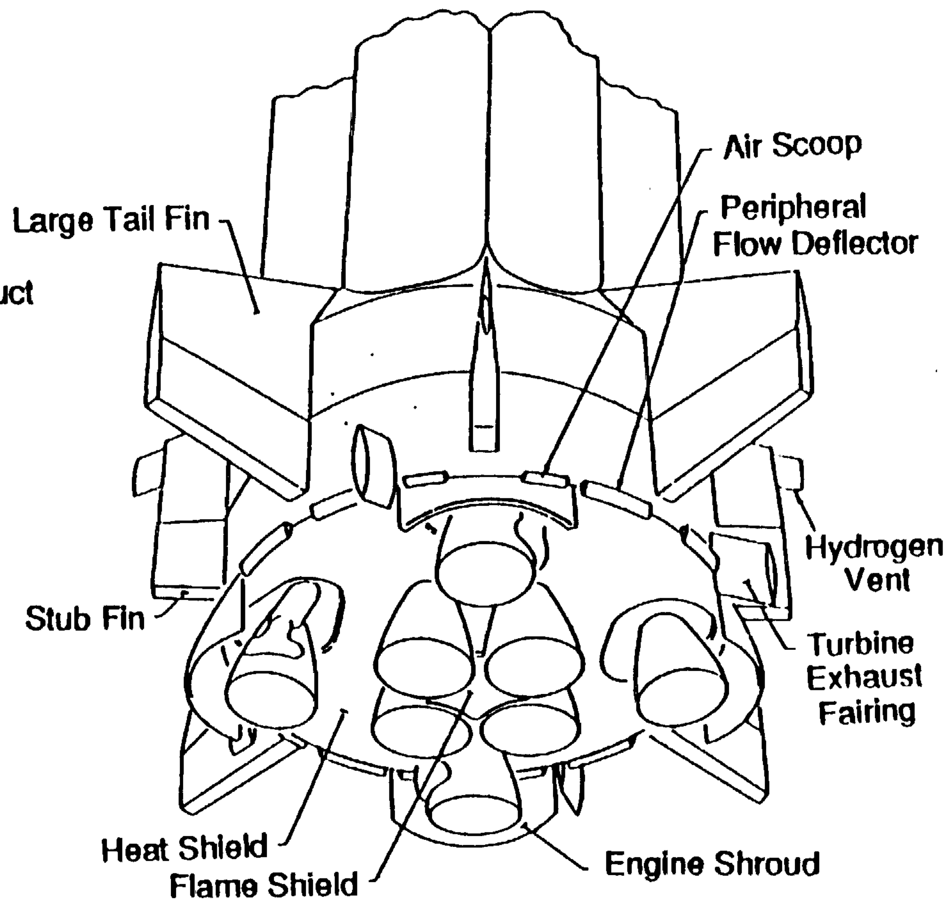


SATURN V/S-1C STAGE - F-1 ENGINE

SCOOPS, FLOW DEFLECTORS, AND TURBINE EXHAUST DUCTS

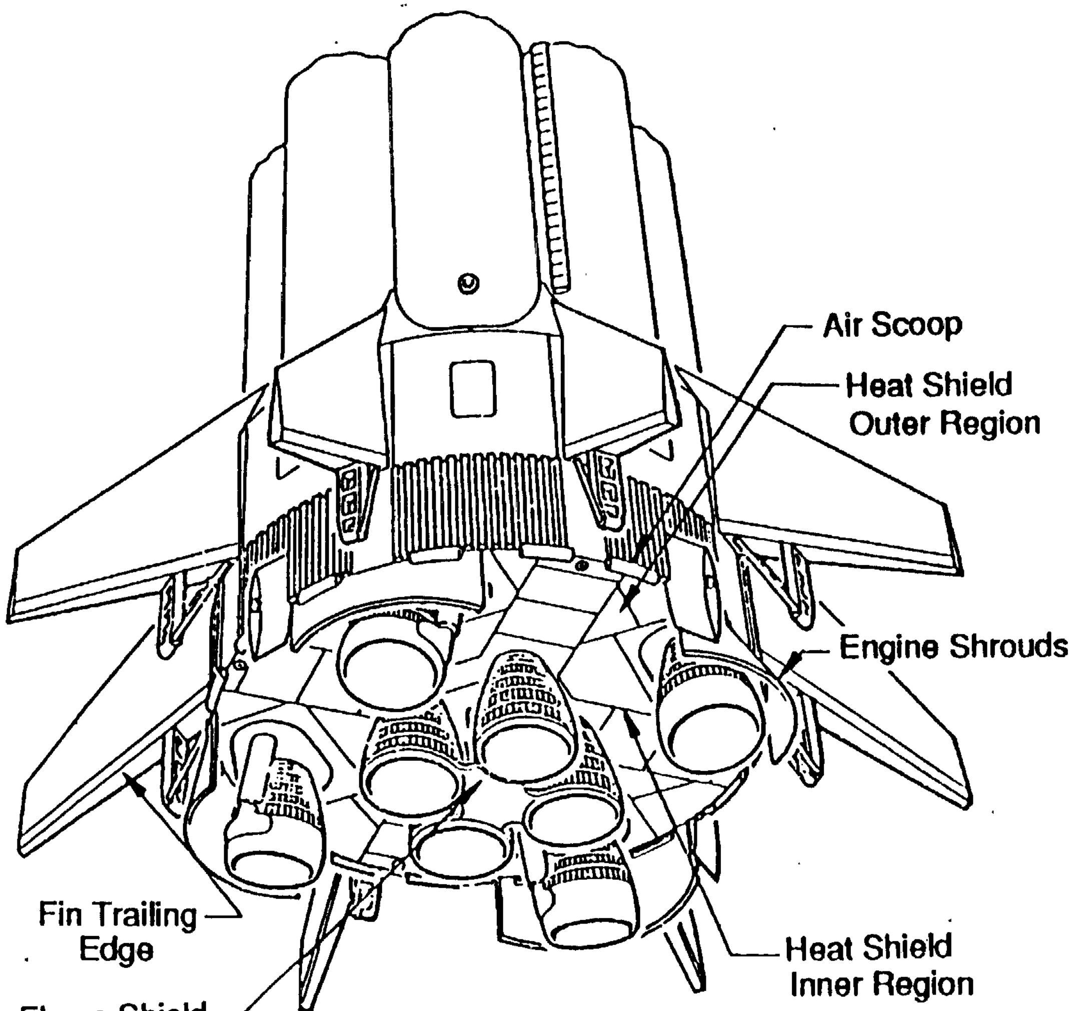


SATURN I, BLOCK I

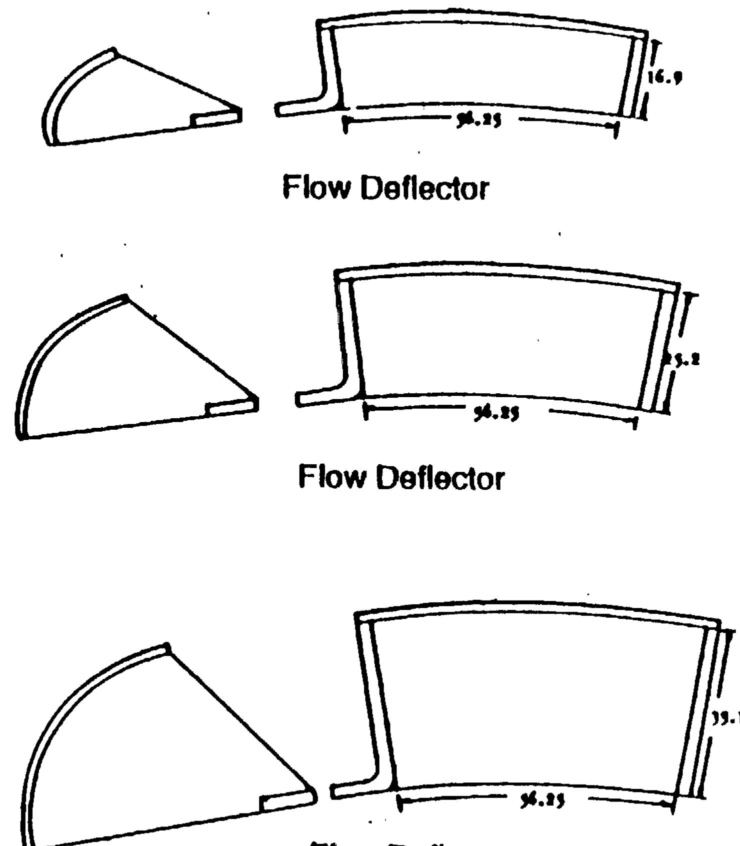


SATURN I, BLOCK II

SCOOPS, FLOW DEFLECTORS, AND TURBINE EXHAUST DUCTS

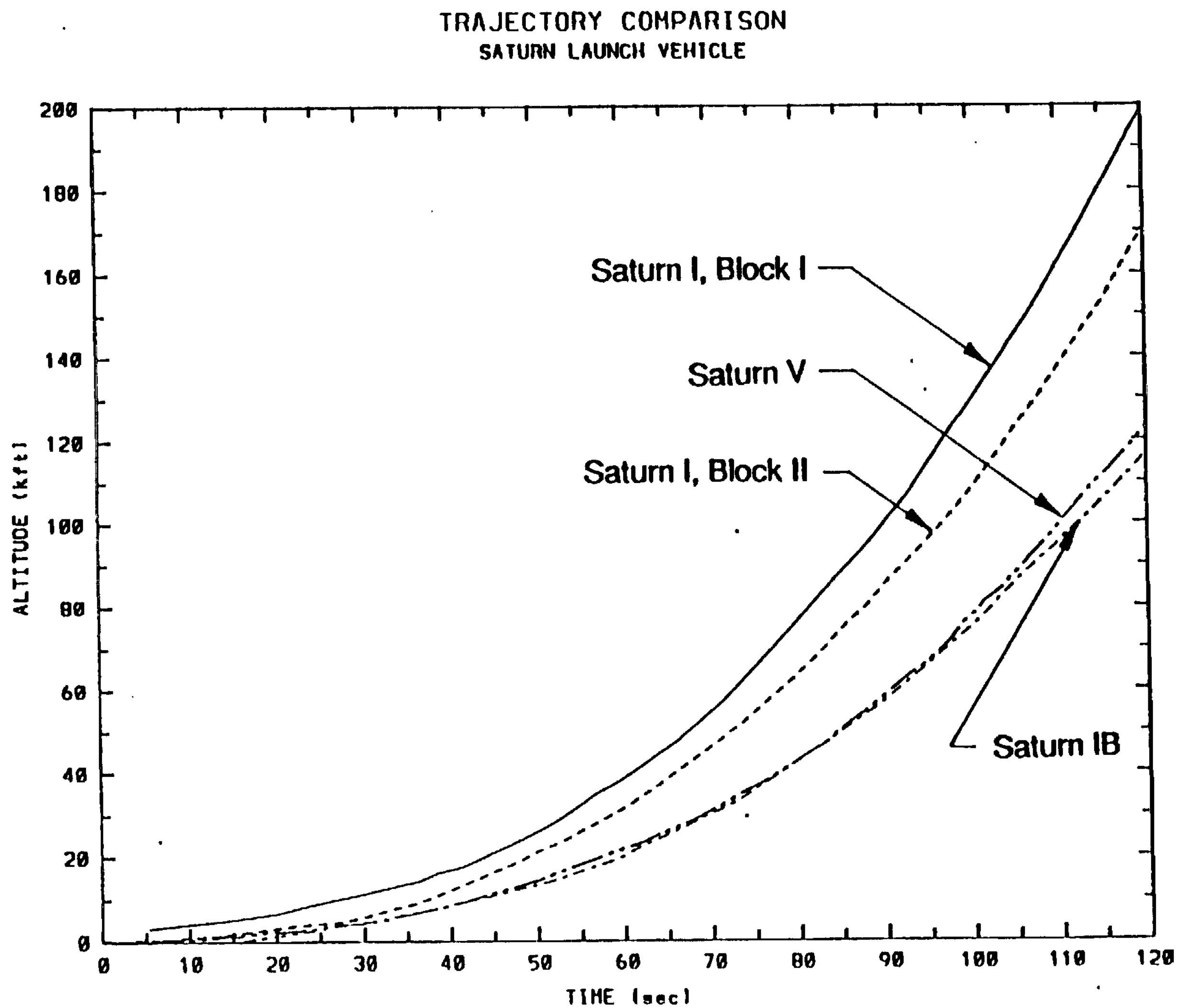


SATURN IB



SATURN V FLOW DEFLECTORS

TYPICAL SATURN FLIGHT VEHICLE TRAJECTORIES



SATURN FLIGHT BASE HEATING INSTRUMENTATION SUMMARY

REMTECH

VEHICLE	FLIGHT	BASE HEAT SHIELD			ENGINES		
		TOT CAL	RAD	GTP	TOT CAL	RAD	GTP
SAT I BK I	SA-1	3	2	5	0	0	0
	SA-2	3	2	5	0	0	0
	SA-4	3	2	5	0	0	0
	SA-4	3	2	5	0	0	0
SAT I BKII	SA-5	5	5	7	0	0	0
	SA-6	5	5	7	0	0	0
	SA-7	5	5	7	0	0	0
	SA-8	5	5	7	0	0	0
	SA-9	5	5	7	0	0	0
	SA-10	5	5	7	0	0	0
SAT IB	AS-201	3	2	2	3	0	0
	AS-202	3	2	2	0	0	0
	AS-203	1	0	0	3	0	0
	AS-204	3	3	3	5	0	1
SAT V	AS-501	5	3	6	12	3	7
	AS-502	5	3	6	12	3	7
	AS-503	5	3	6	12	3	7
	AS-504	4	3	6	12	3	1
	AS-505	4	3	6	12	3	1
	AS-506	2	0	2	0	0	0
	AS-507	2	0	2	0	0	0
	AS-508	2	0	2	0	0	0
	AS-509	2	0	2	0	0	0
	AS-510	2	0	2	0	0	0
	AS-511	2	0	2	0	0	0
	AS-512	2	0	2	0	0	0

SATURN FLIGHT BASE HEATING
INSTRUMENTATION GENERAL DESCRIPTIONS

INSTRUMENT	COMMENTS
Total Calorimeters	•Slug type, Saturn I, Bk I & Bk II •Membrane, Saturn IB & Saturn V •Typical ranges: 0 - 40, 0 - 100 BFS — Sat . I & IB 0 - 40, 0 - 60, 0 - 100 BFS — Sat. V
Radiometers	•Slug type with sapphire window, Saturn I, Bk I & Bk II •Membrane with sapphire window, Saturn IB •Membrane with sapphire window and nitrogen purge, Sat. V •Typical ranges: 0 - 40 BFS, Sat. I & IB H.S. 0 - 100, BFS, Sat. I & 1B F.S. 0 - 20 BFS, Sat. V H.S. 0 - 60, 0 - 100 BFS, Sat. V Engine
Gas Temperature Probes	•Unshielded, single & double shield T/C - Sat. I, Bk I & II •Exposed T/C with Guard Ring, Sat. IB •Double Shielded (Platinum) T/C, Sat. V •Typical ranges: 0 - 1500, 0 - 1750, 0 - 2000°C Sat. I & IB 0 - 1750°C, Sat. V

SATURN FLIGHT BASE HEATING INSTRUMENTATION ACCURACY AND DATA QUALITY

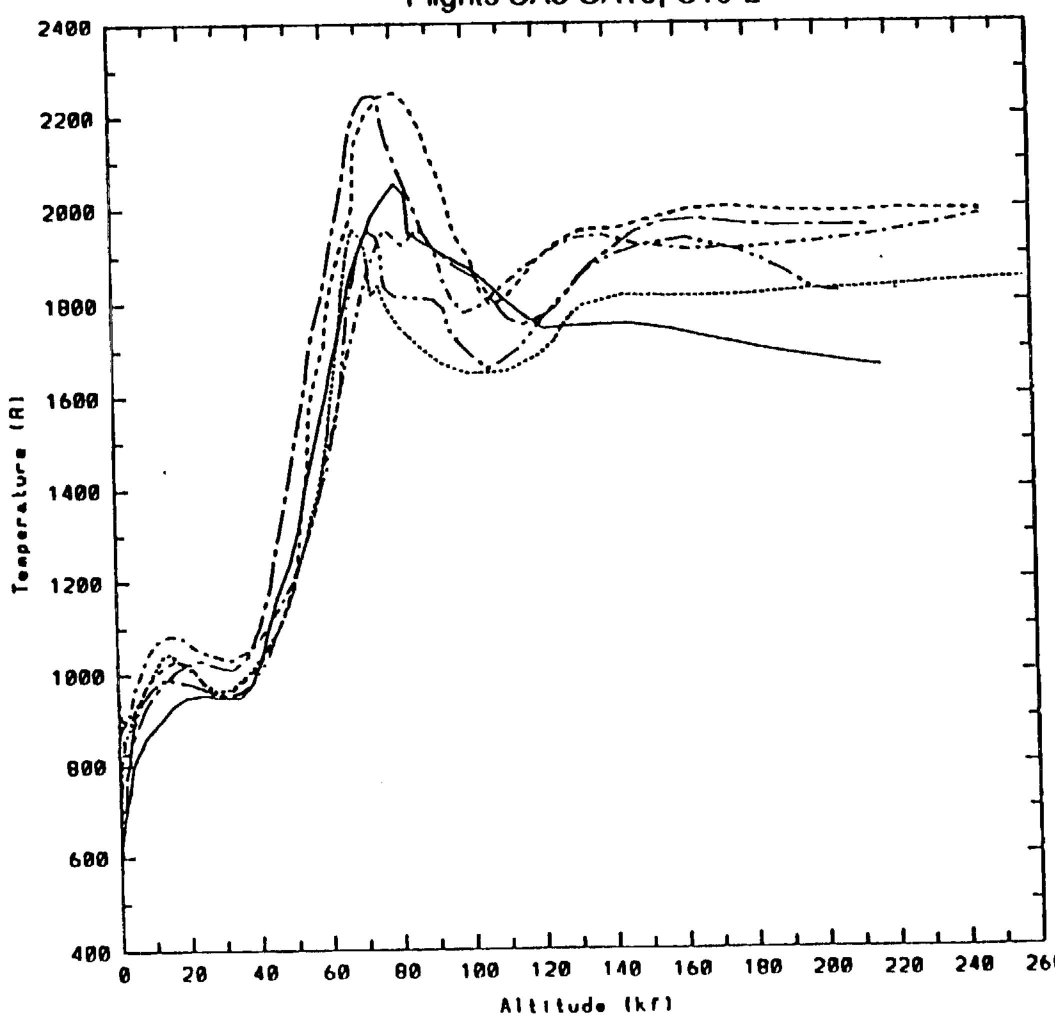
- ***ACCURACY***
  - Gage error as delivered usually 3 to 5% of full scale
  - Gage accuracy when installed and subjected to flight conditions very difficult to define and different for each gage location

- ***DATA QUALITY***
  - Very difficult to assess quantitativly because of:
    - Instrument groupings and local flow effects
    - Radiation and convection losses
    - Surface emissivity and sensor element absorptivity
    - Mounting positions and installation features
    - Thermal boundary discontinuity
    - Directional sensitivity
    - Operational flight influences
      - Water cooling at liftoff
      - Venting
      - Surface ablation and sensor contamination

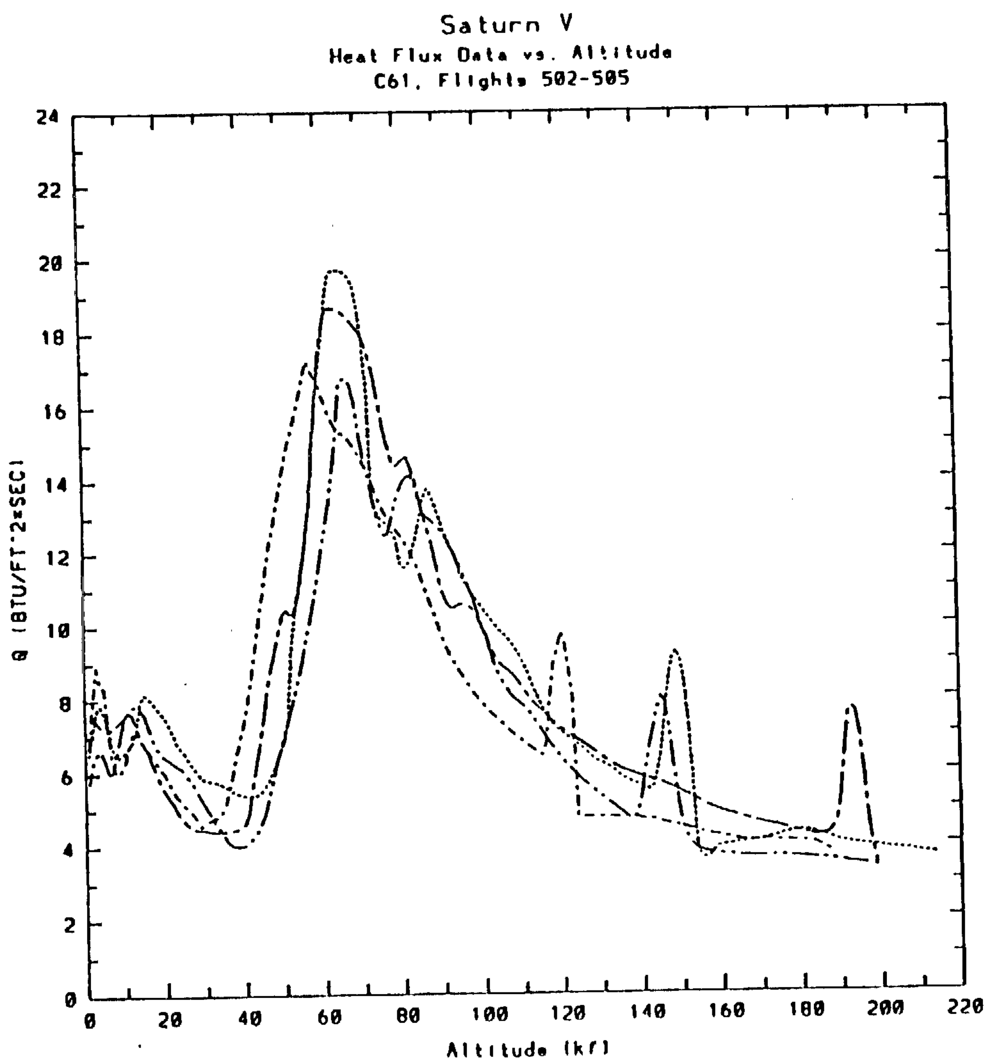
- ***References for Detailed Studies of Instrument Performance:***
  - Teledyne Brown Report EE-MSFC-1774, Volumes I and II, December 1973
  - NASA-MSFC TMX-53326, September 1965
  - Boeing Company Report 5-9410-H-448, September 1972
  - NASA CR-61390, May 1972

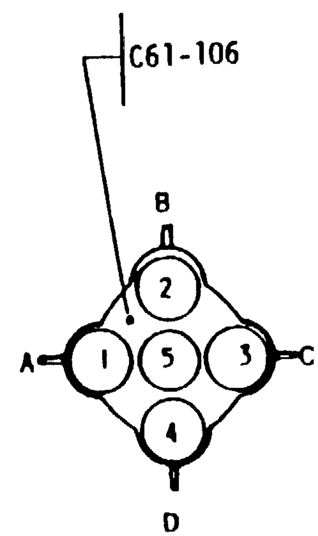
SATURN FLIGHT BASE HEATING DATA REPEATABILITY

Saturn I Block II
Gas Temperature Data vs. Altitude
Flights SA5-SA10, C10-2

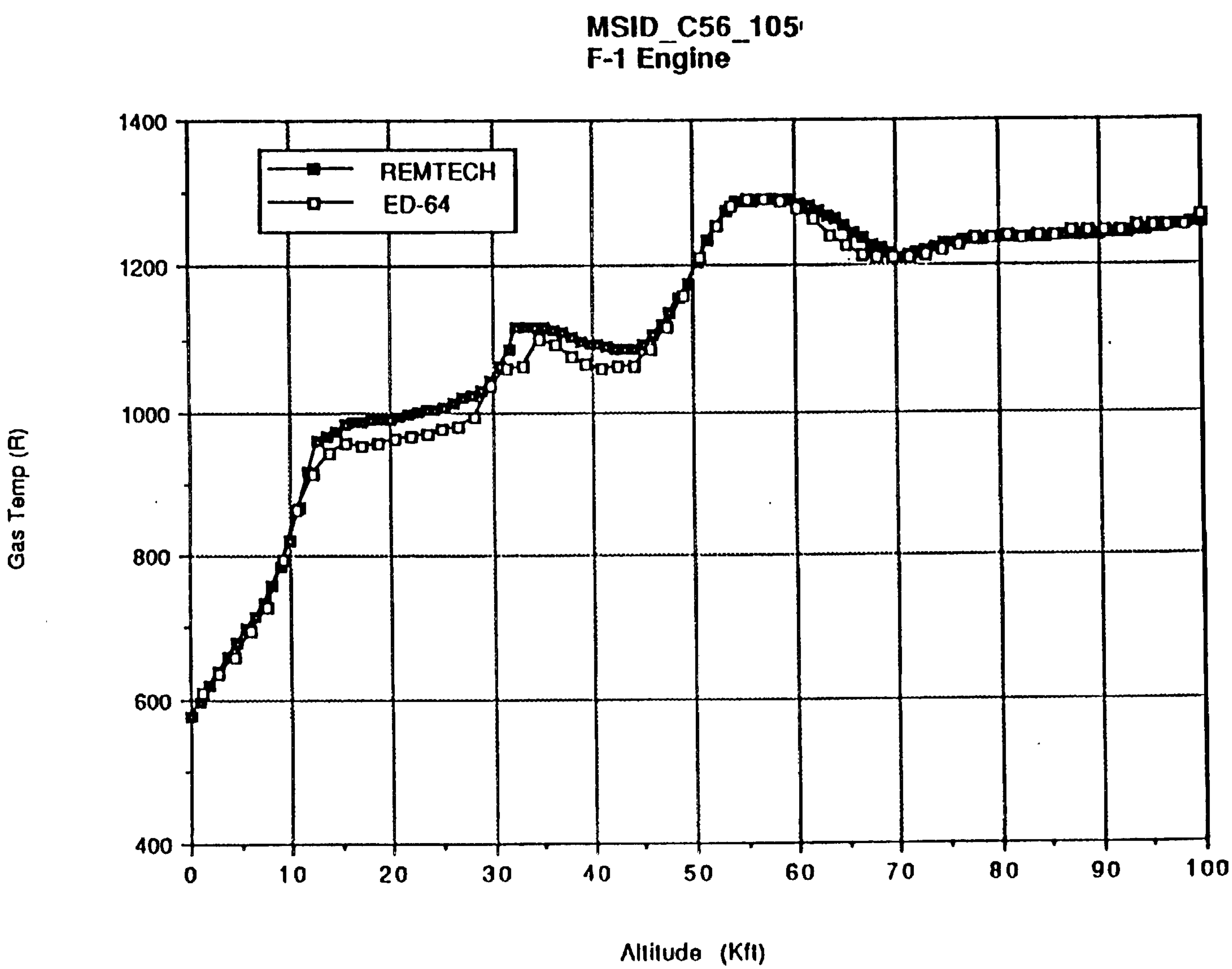


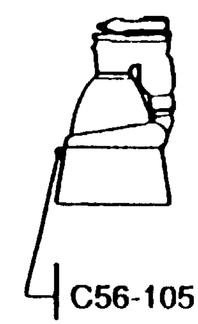
SATURN FLIGHT BASE HEATING DATA REPEATABILITY



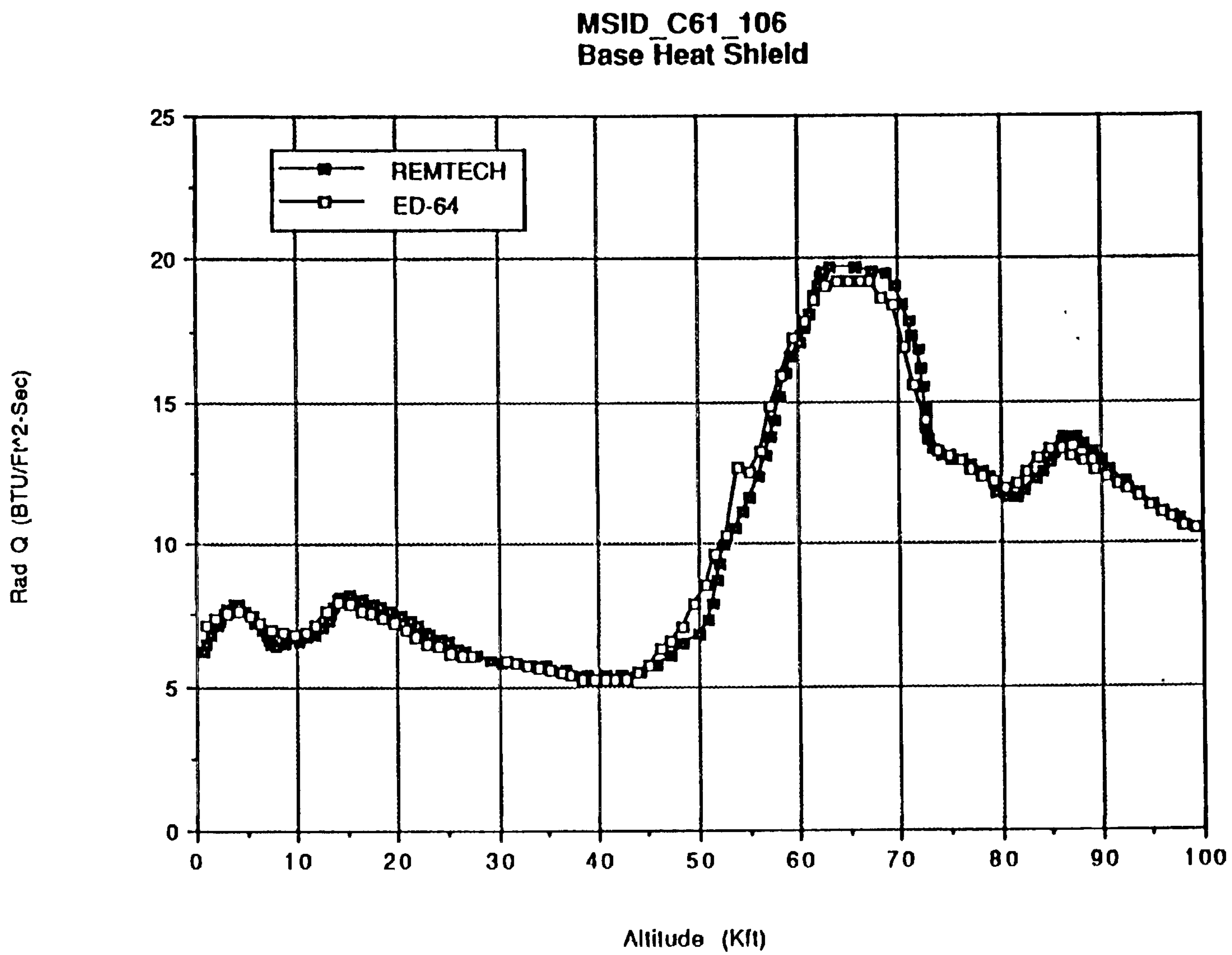


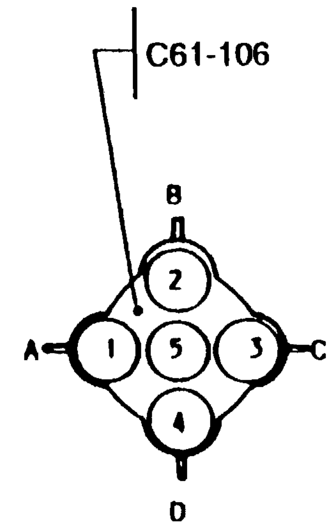
COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64





COMPARISON OF SATURN V FLIGHT DATABASES
REMTECH vs MSFC ED-64





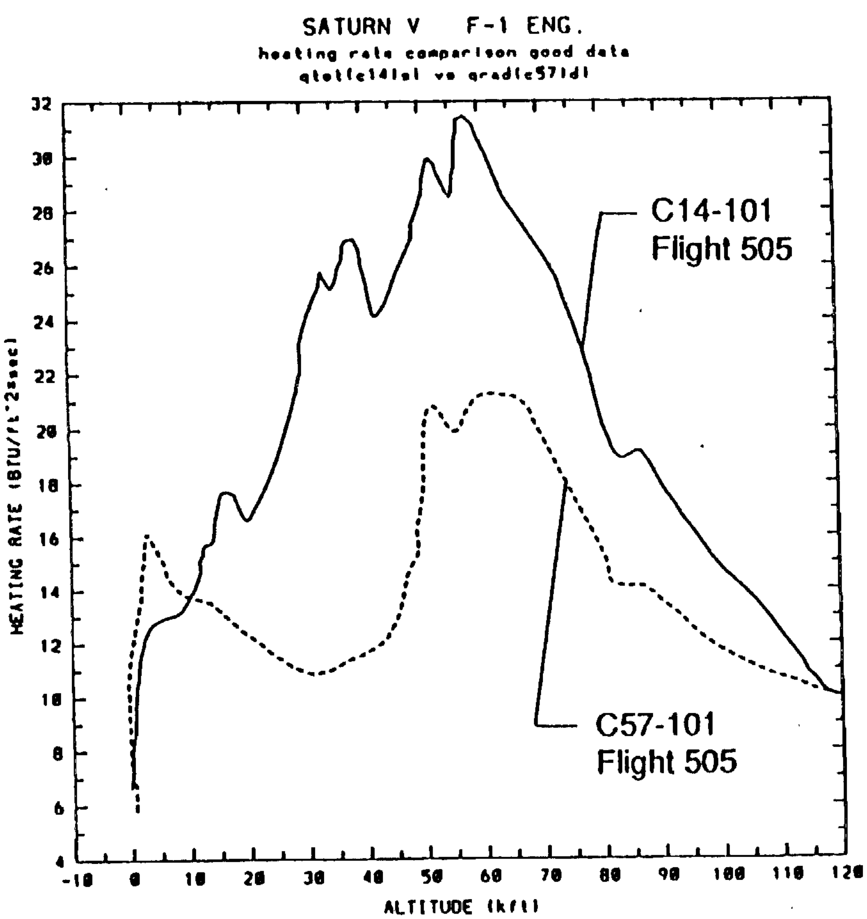
OBJECTIVE 1
SATURN FLIGHT DATA REVIEW
TO DEFINE h_c

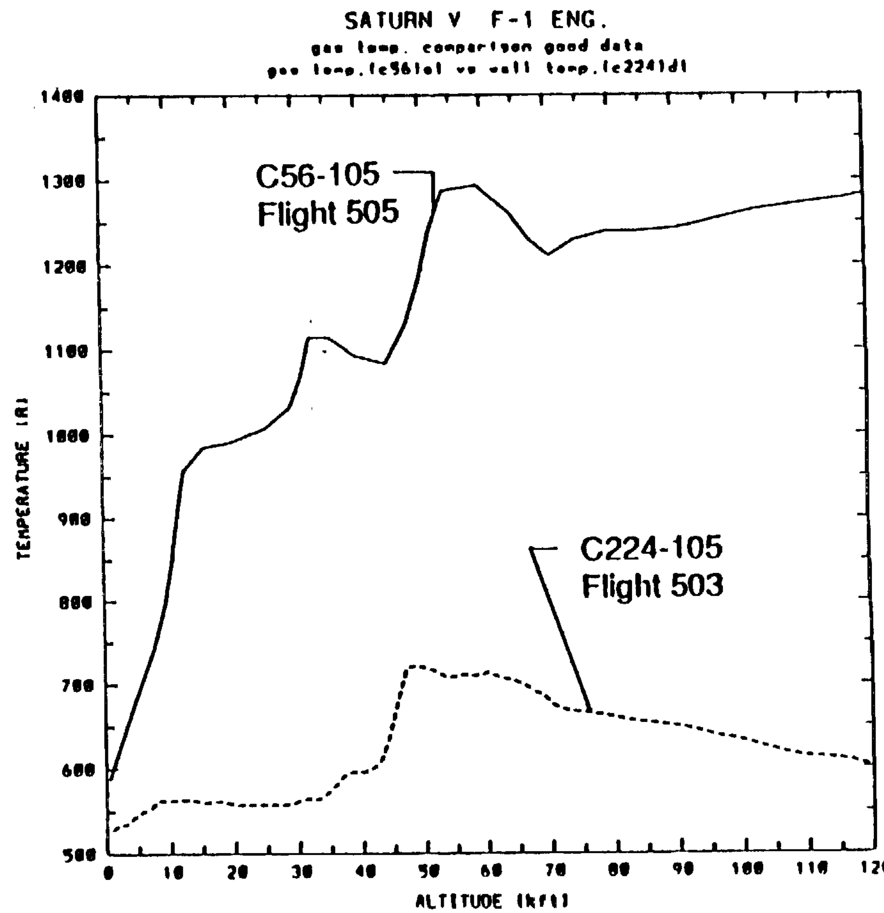
INSTRUMENT GROUPING FOR h_c REDUCTION

$$h_c = \frac{Q_{Tot} - Q_{Rad}}{T_{Gas} - T_{Wall}}$$

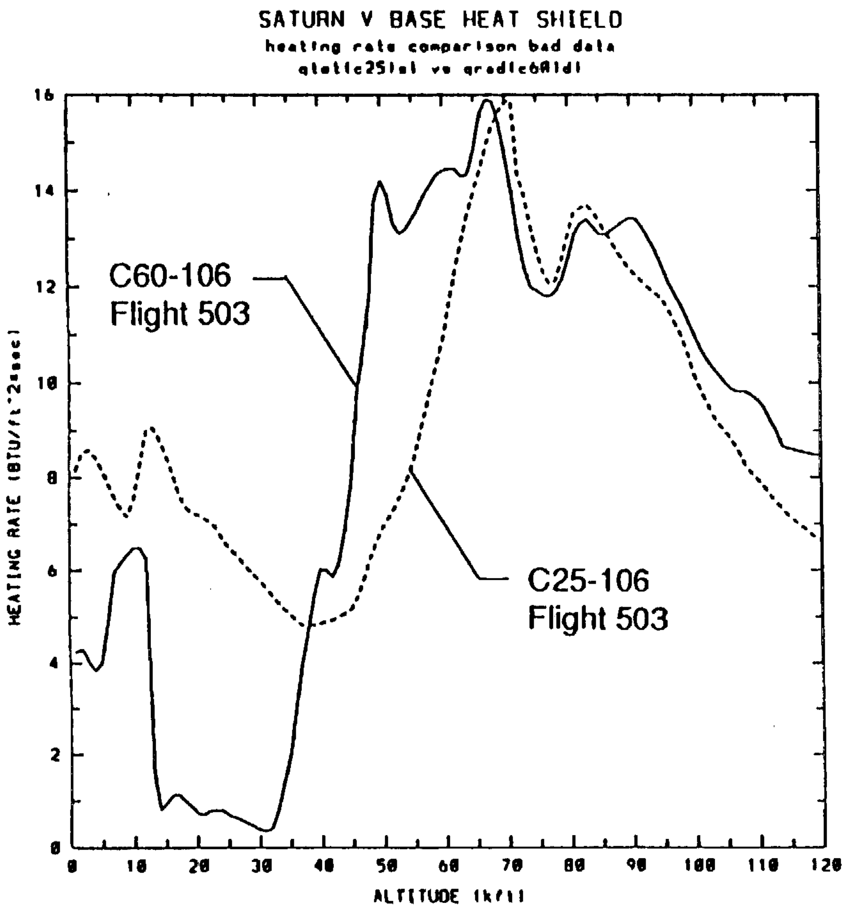
VEHICLE		INSTRUMENT			
		TOT CAL	RAD	GTP	T_{WALL}
SATURN I BK I HEAT SHIELD		C76-3 C63-1	C79-2 C64-4	C65-3 C65-3	C76-3(wall) Flight SA-4 C63-1(wall) FLight SA-4
SATURN I BK I HEAT SHIELD		C76-3 C63-1 C194-1 C194-1	C193-8 C190-5 C192-2 C189-4	C65-3 C196-5 C198-6 C10-4	C76-3 (wall) SA-4 C63-1 (wall) SA-4 C76-3 C76-3 (wall) SA-4
SATURN IB	Heat Shield	C611-3 C508-3	C609-3 C506-7	C610-3 C507-3	C233-106 AS-504 C233-106 AS-504
	H-1 Engine	—	—	—	—
SATURN V	Heat Shield	C25-106 C26-106 C27-106	C60-106 C61-106 C151-106	C49-106 C50-106 C55-106	C233-106 C233-106 C233-106
	F-1 Engine	C14-101 C14-101	C57-101 C57-101	C44-101 C56-105	C224-105 C234-106

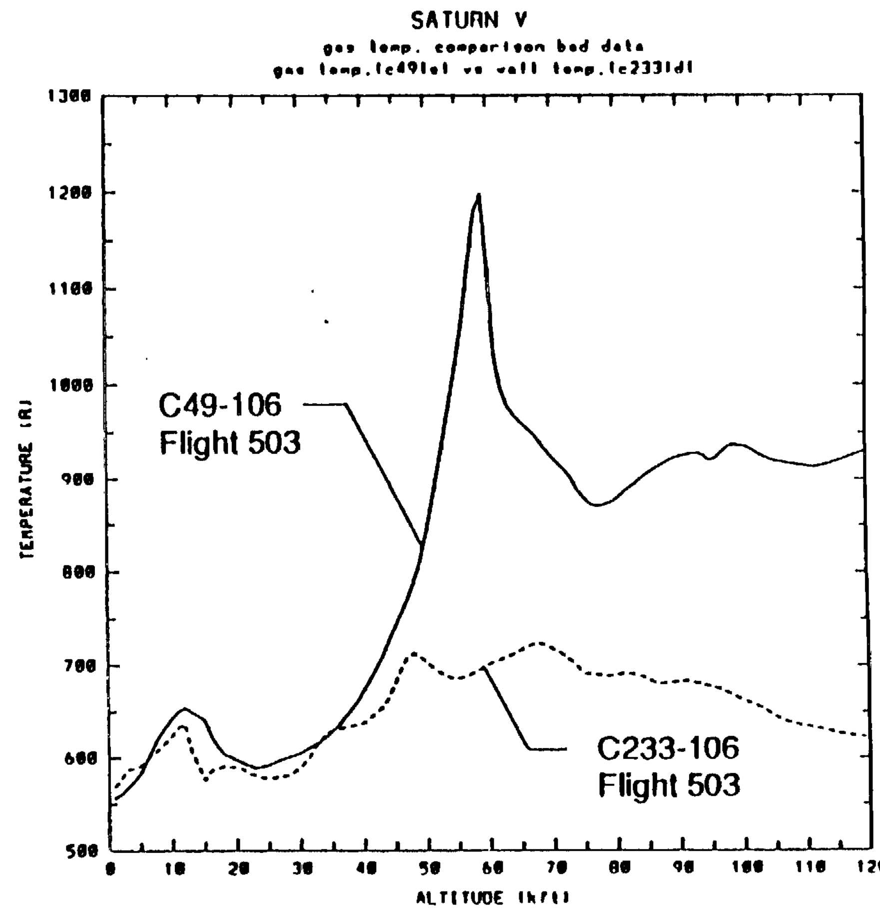
SATURN V TYPICAL "GOOD" BASE HEATING DATA



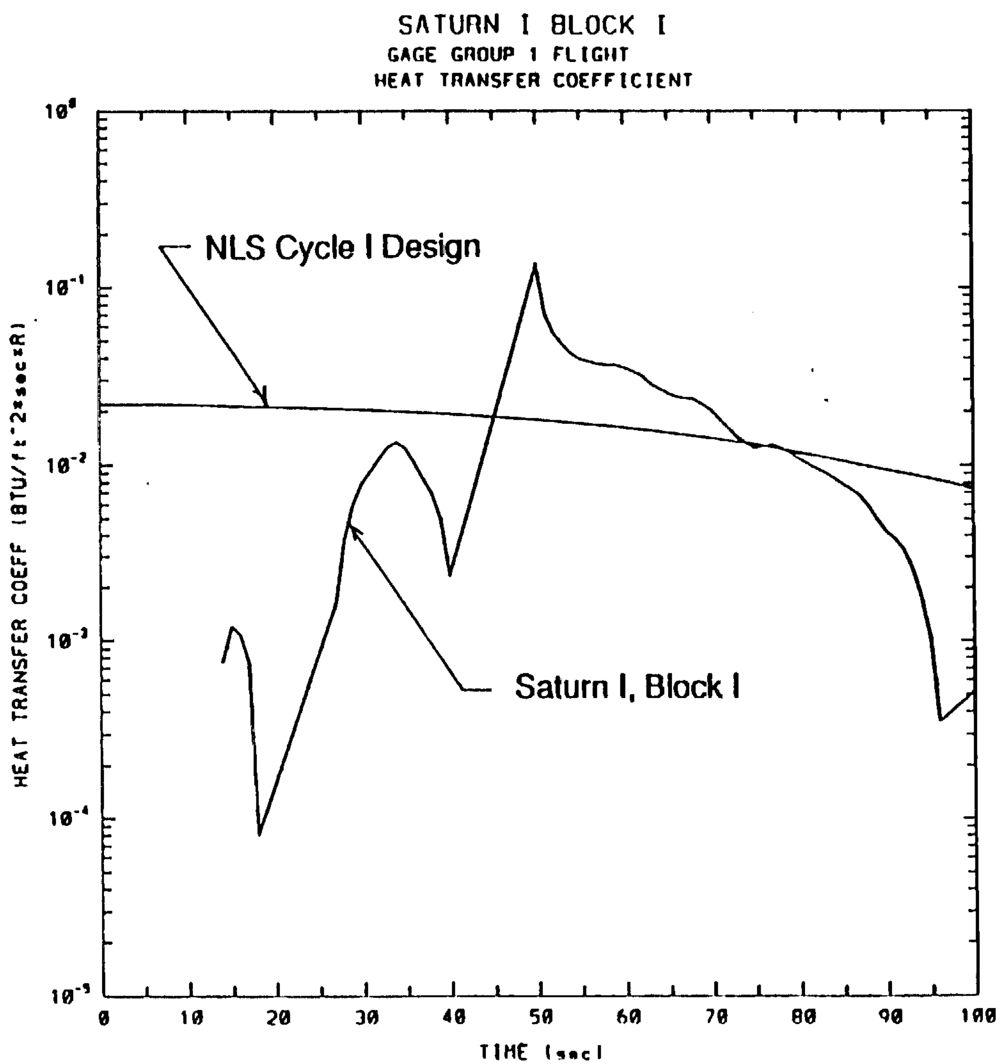


SATURN V TYPICAL "BAD" BASE HEATING DATA

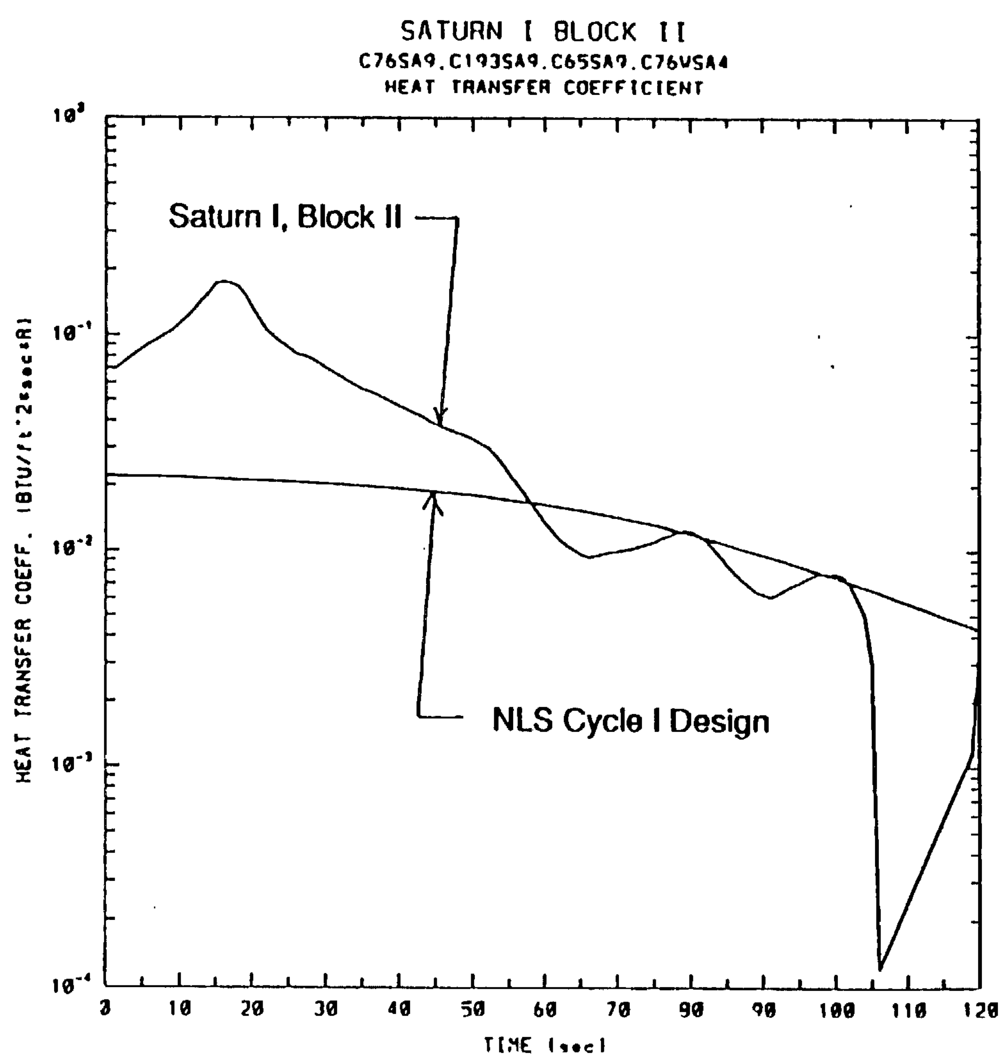




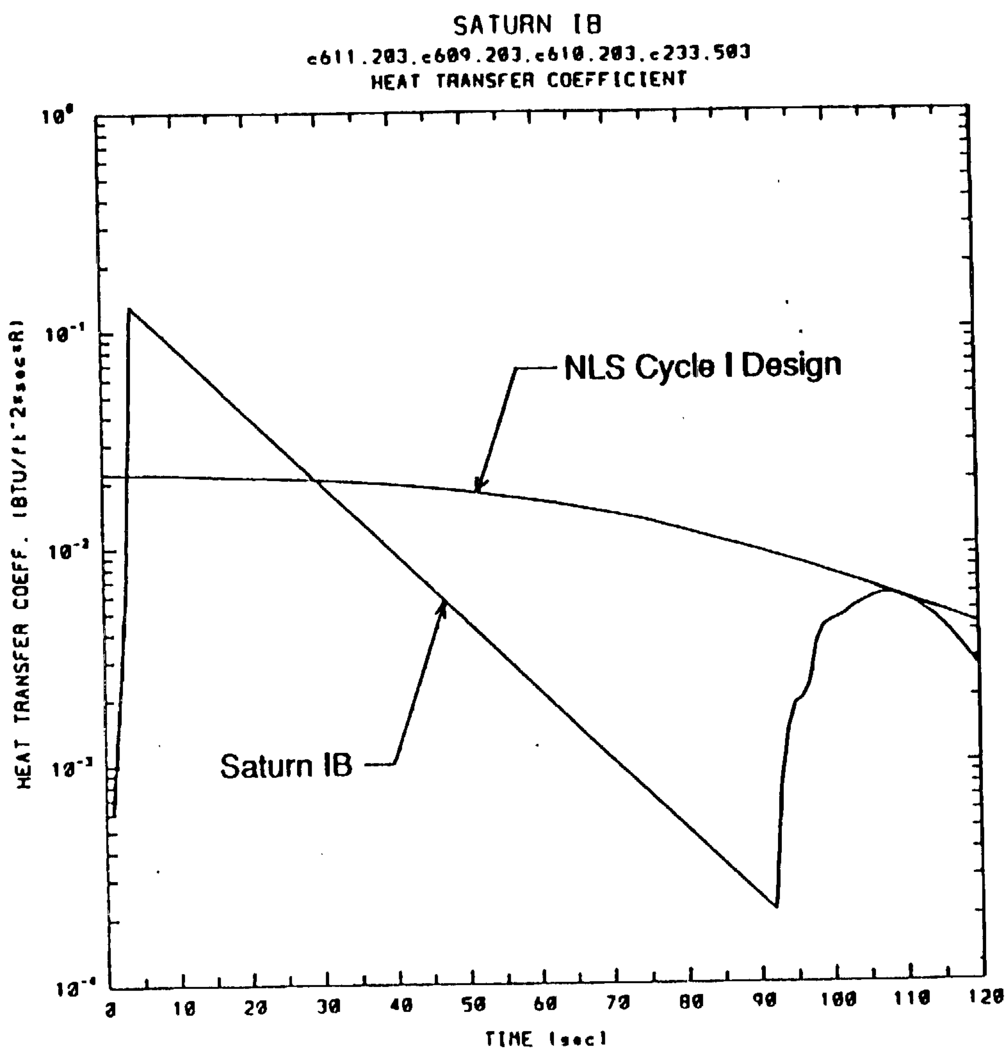
TYPICAL SATURN I BLOCK I FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



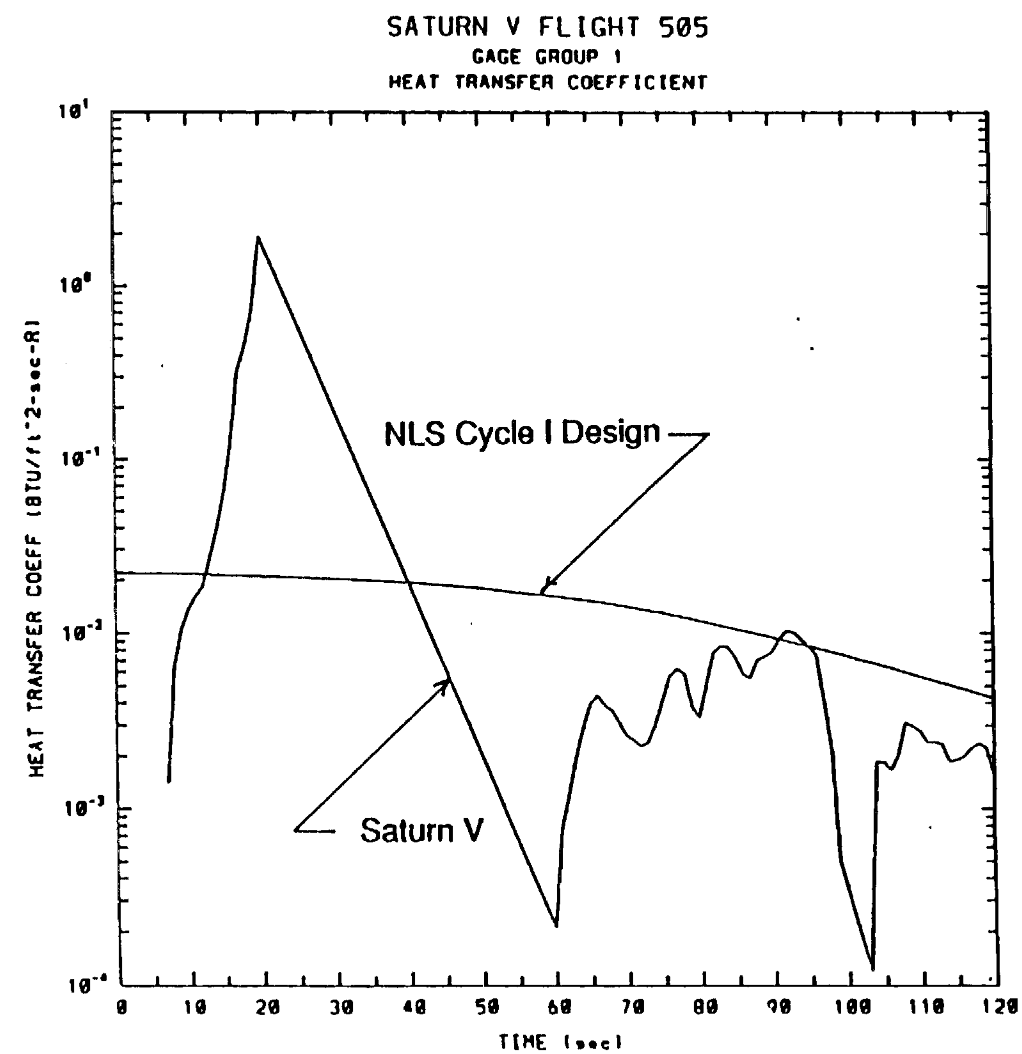
TYPICAL SATURN I BLOCK II FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



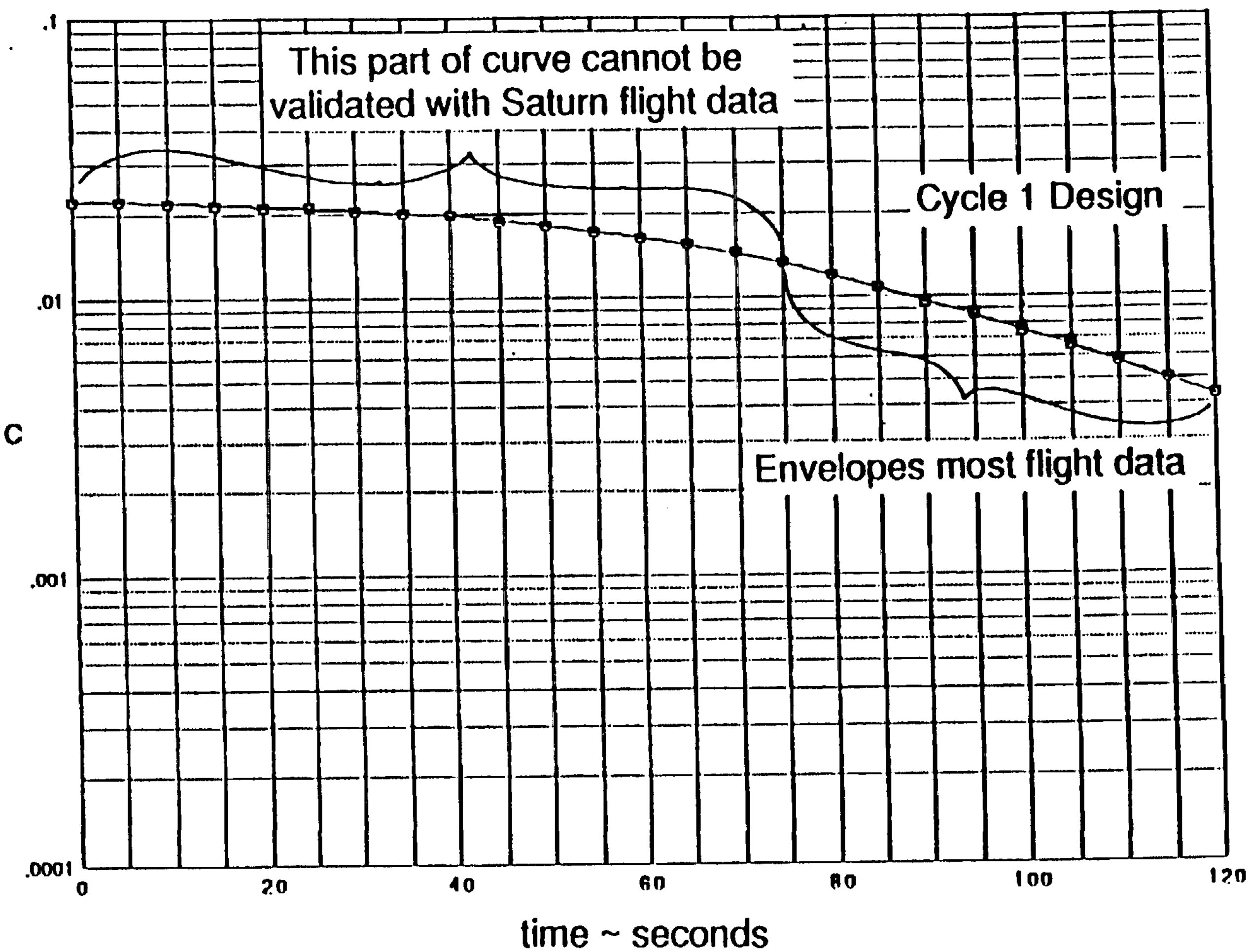
TYPICAL SATURN IB FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



TYPICAL SATURN V FLIGHT DEDUCED CONVECTIVE HEAT TRANSFER COEFFICIENT



RESULTS OF FLIGHT DEDUCED h_C : OBJECTIVE 1



- Above 40,000 Kft, Cycle 1 h_c envelopes Saturn flight data and is valid for NLS 1.5 stage.
- A technique to correlate h_c with base flow conditions for altitudes below 40,000 feet was indicated.

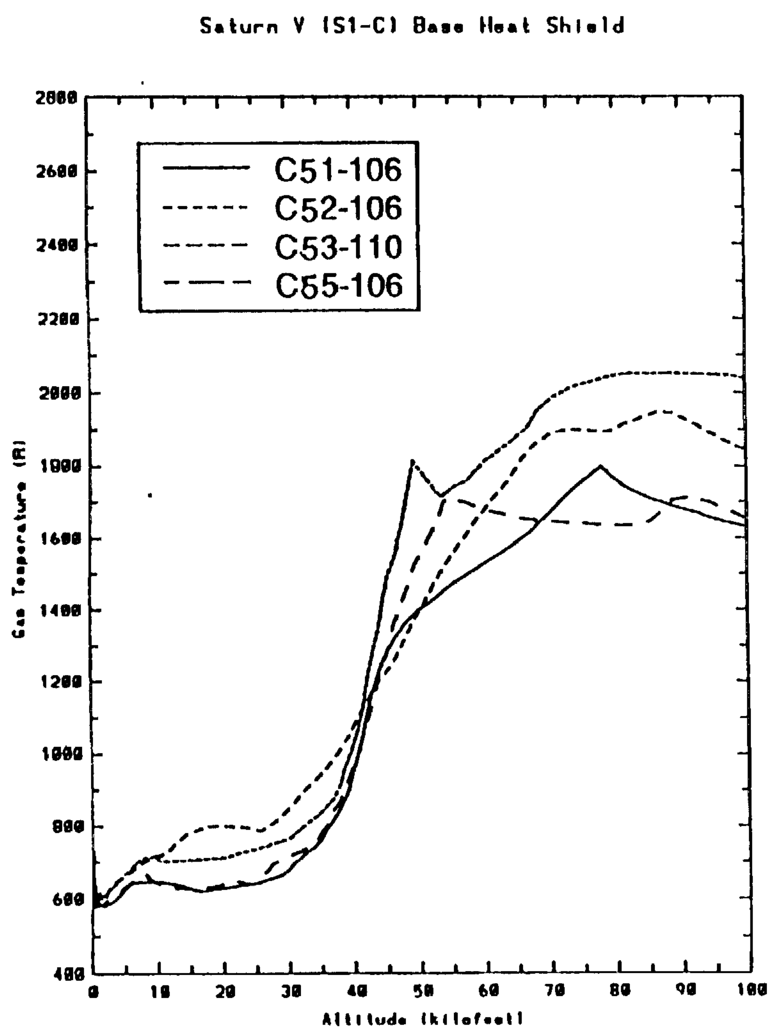
OBJECTIVE 2

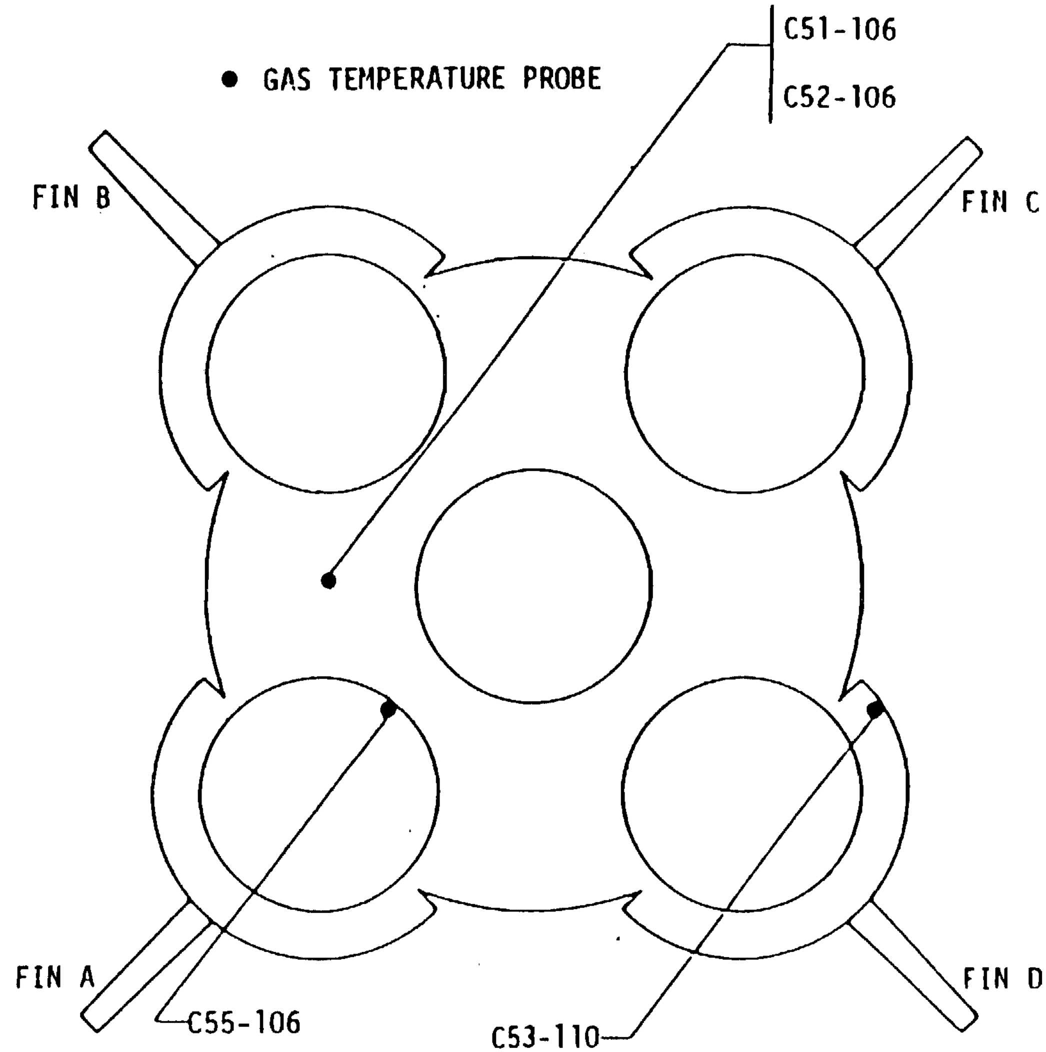
SATURN FLIGHT DATA REVIEW

TO DEFINE T_{gas}

SATURN V GAS TEMPERATURE VARIATION WITH BASE HEAT SHIELD LOCATION

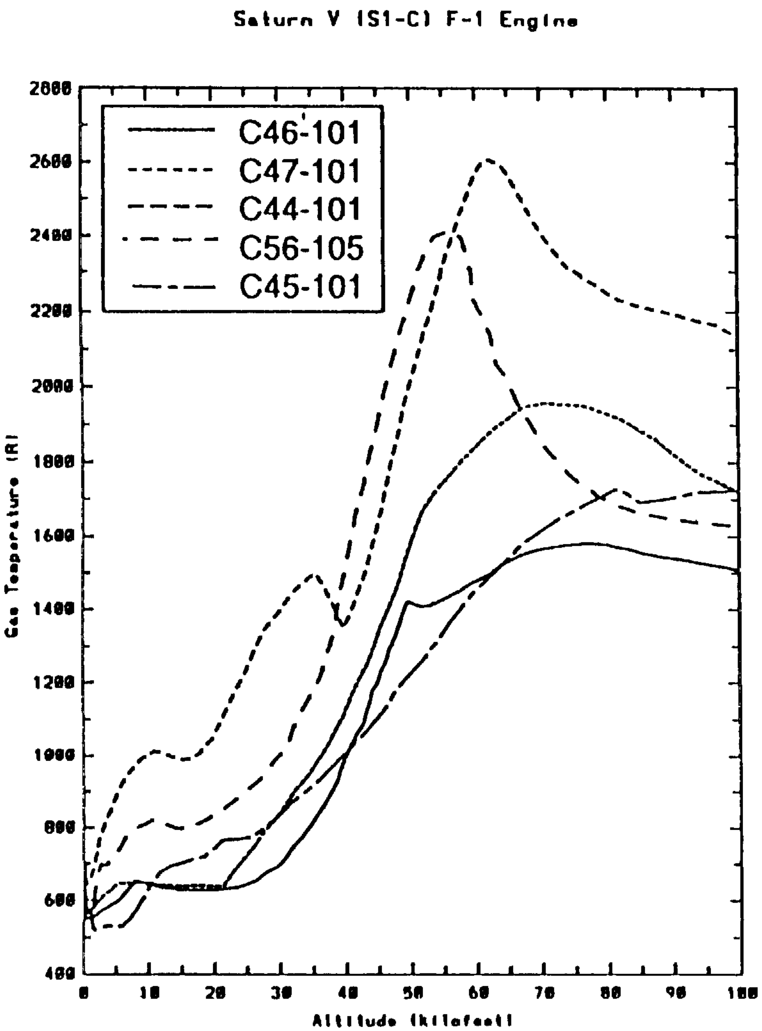
FLIGHT AS-502

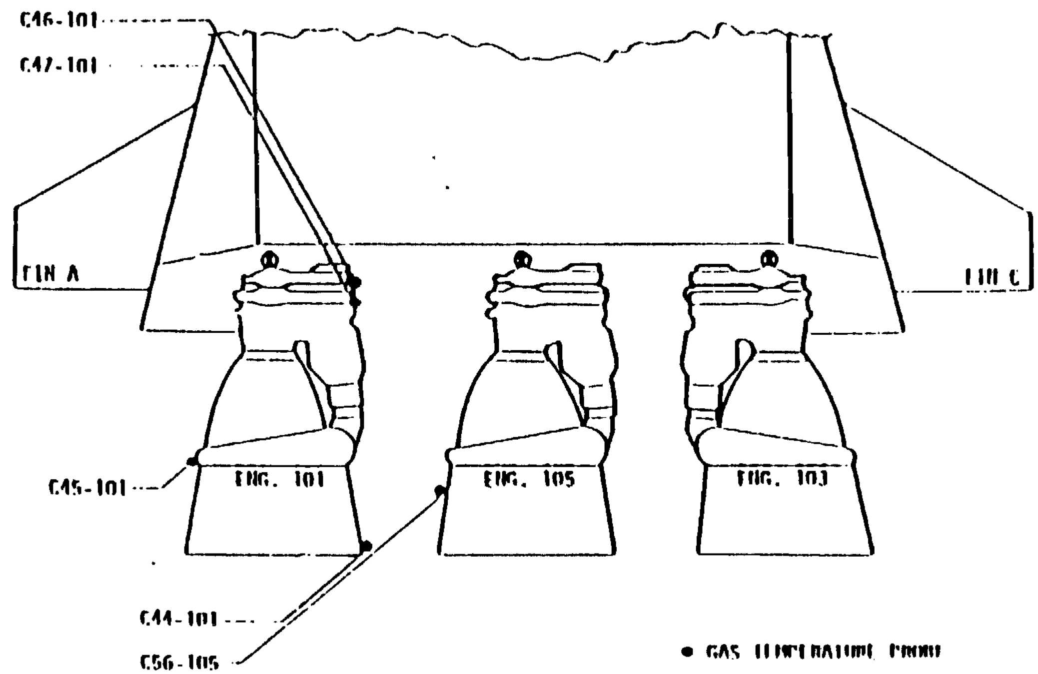




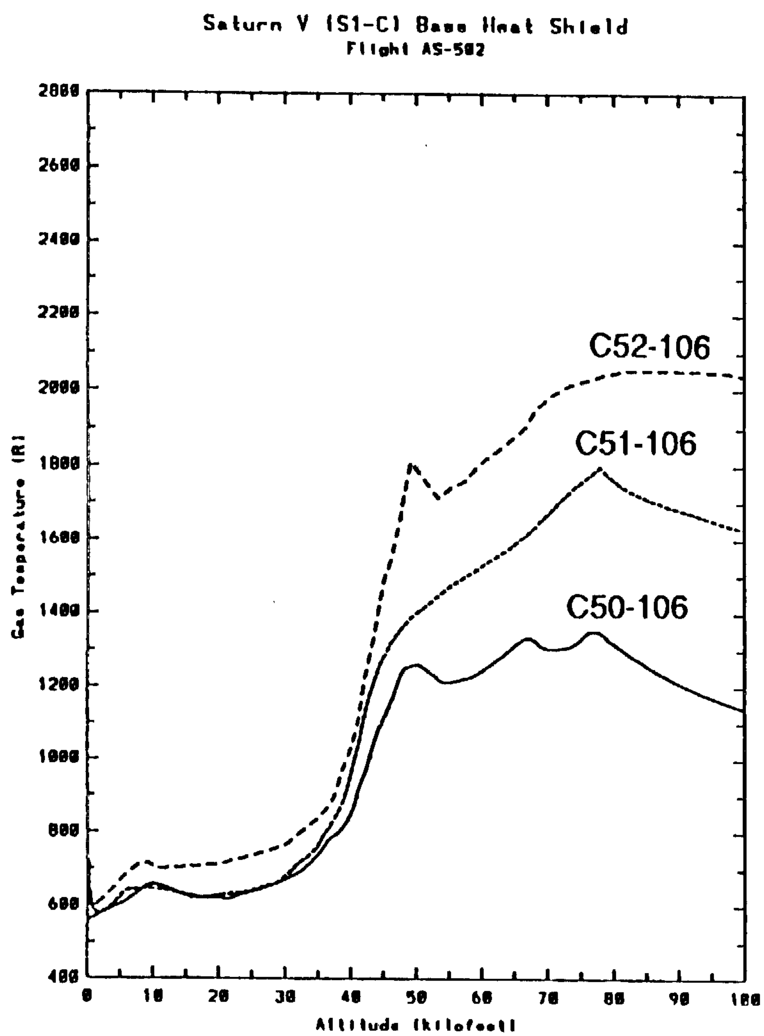
SATURN V GAS TEMPERATURE VARIATION WITH F-1 ENGINE LOCATION

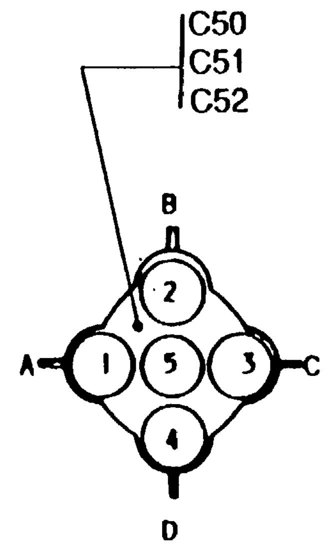
FLIGHT AS-502





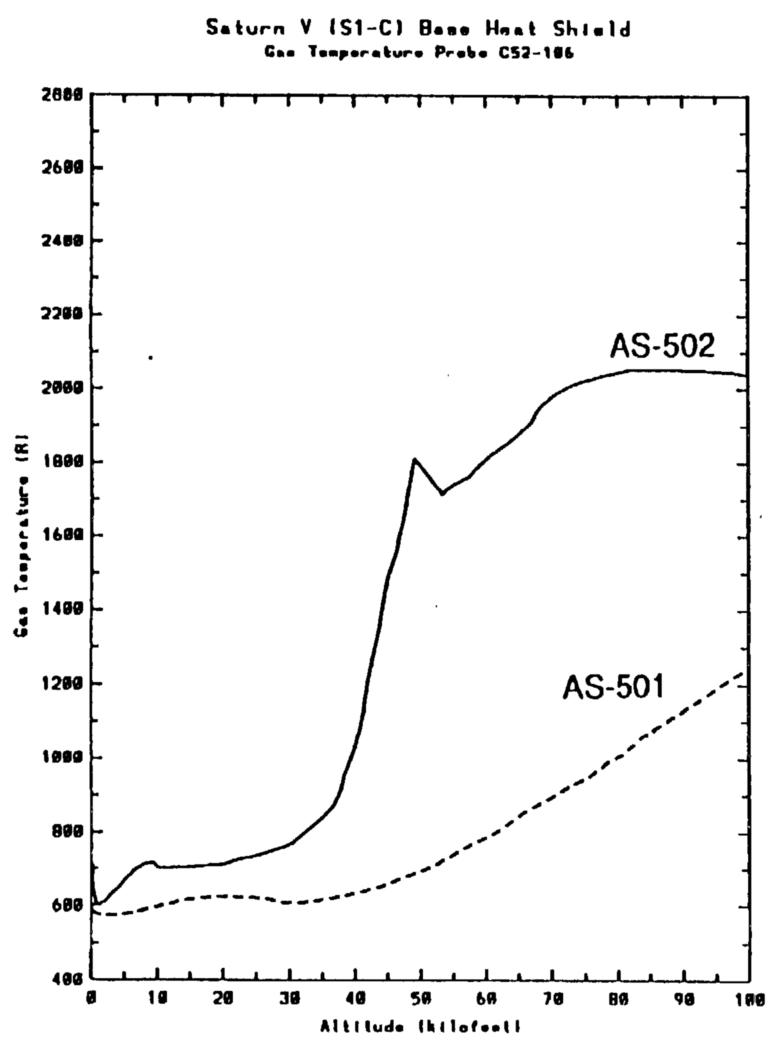
SATURN V GAS TEMPERATURE EFFECT OF PROBE HEIGHT OFF HEAT SHIELD SURFACE

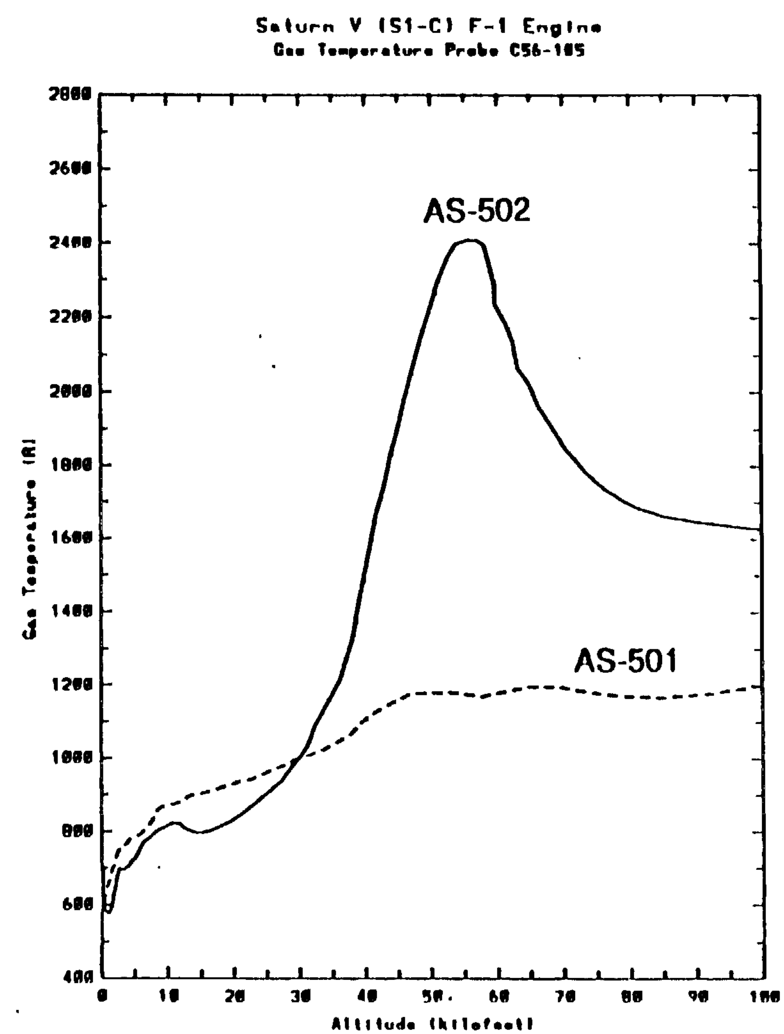




Probe	Height Off Surface
C50	0.25"
C51	1.00"
C52	2.50"

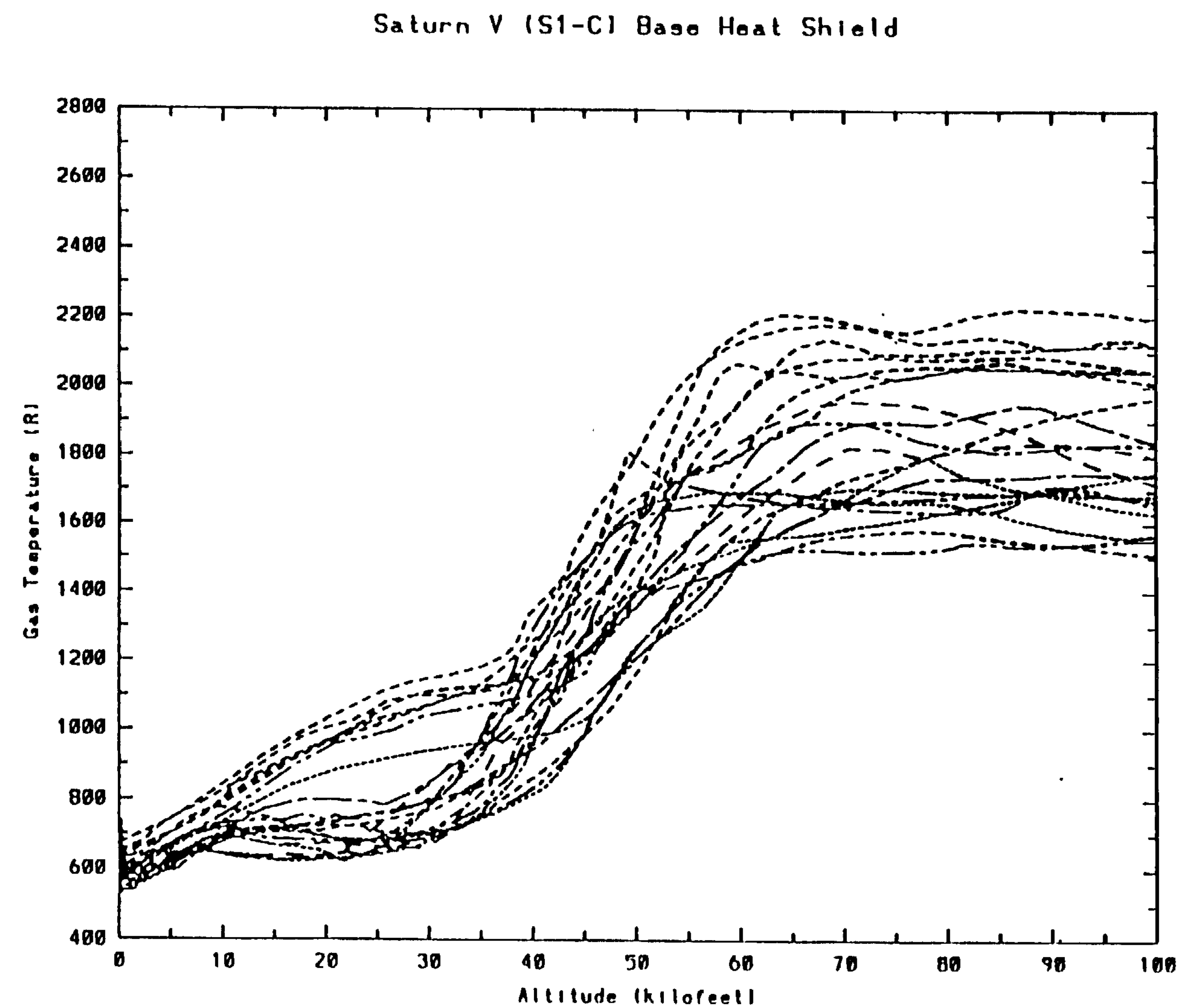
SATURN V GAS TEMPERATURE FLOW DEFLECTOR EFFECT
AS-501 vs AS-502





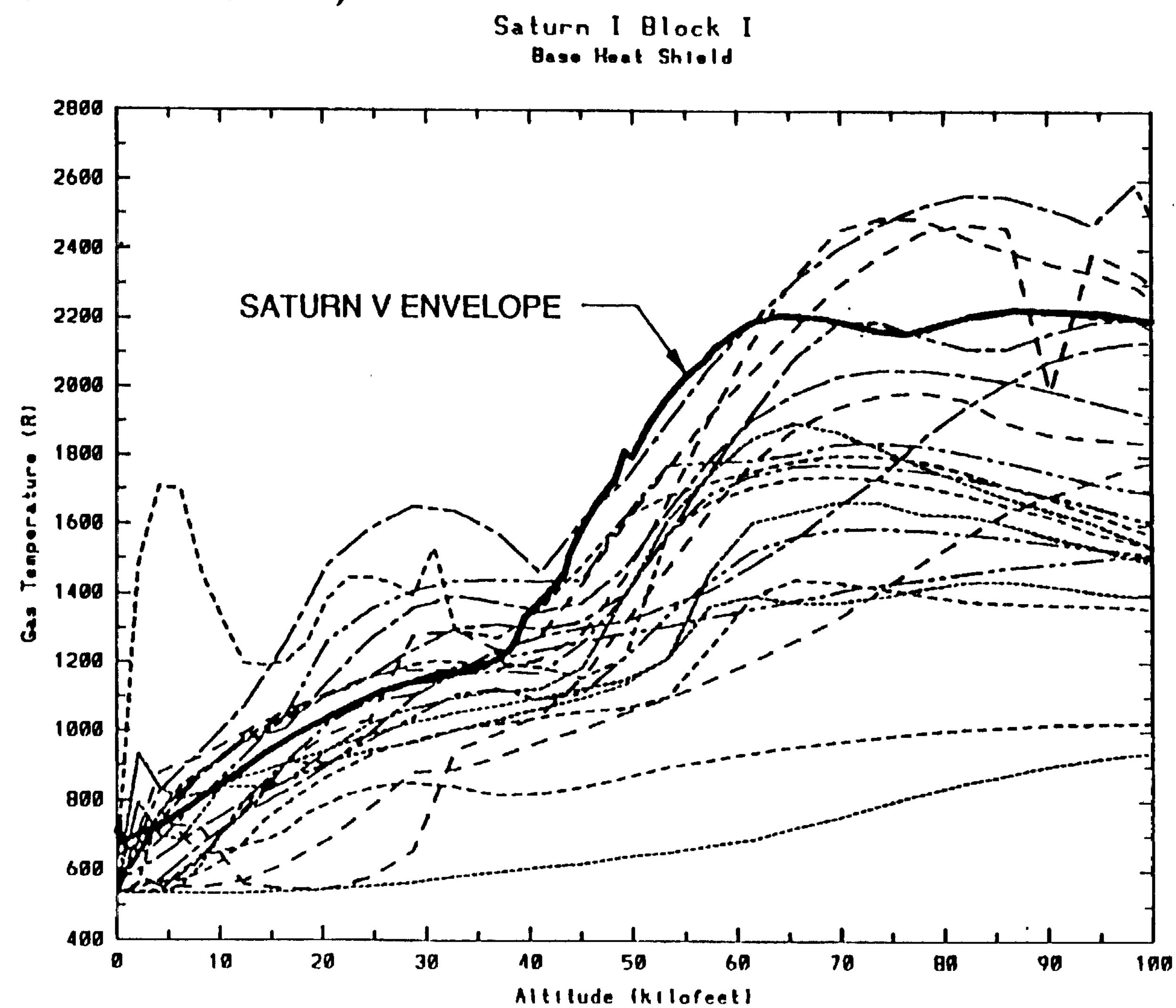
SATURN V GAS TEMPERATURE BASE HEAT SHIELD DATA FLIGHTS AS-502 - AS-509

22 FLIGHT MEASUREMENTS



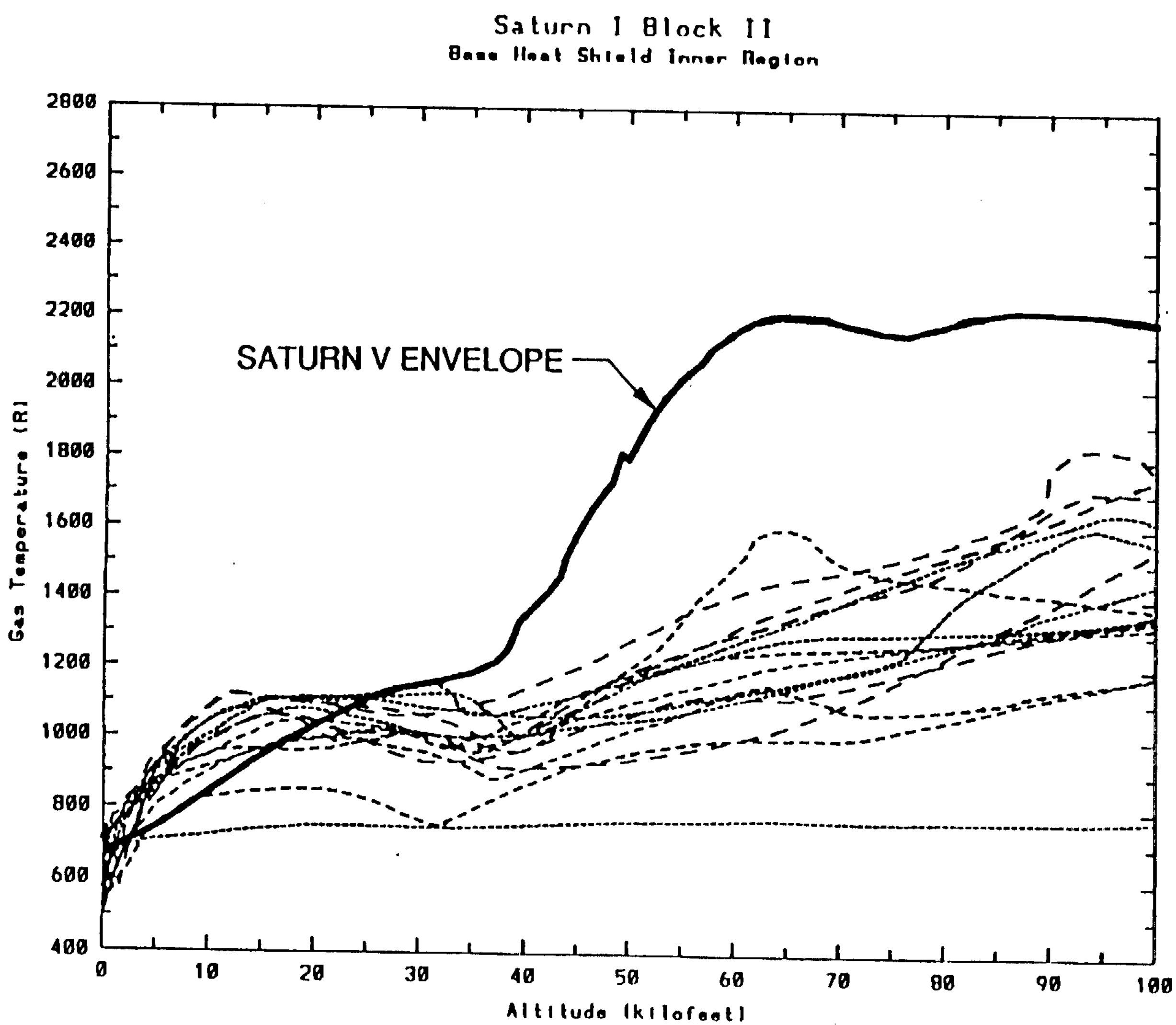
SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK I

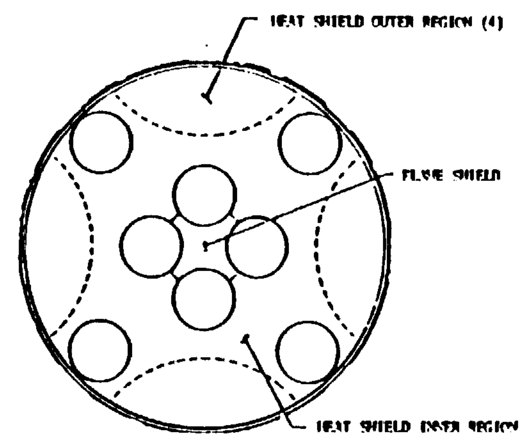
19 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK I
(Does not Include Flame Shield)



SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK II INNER REGION

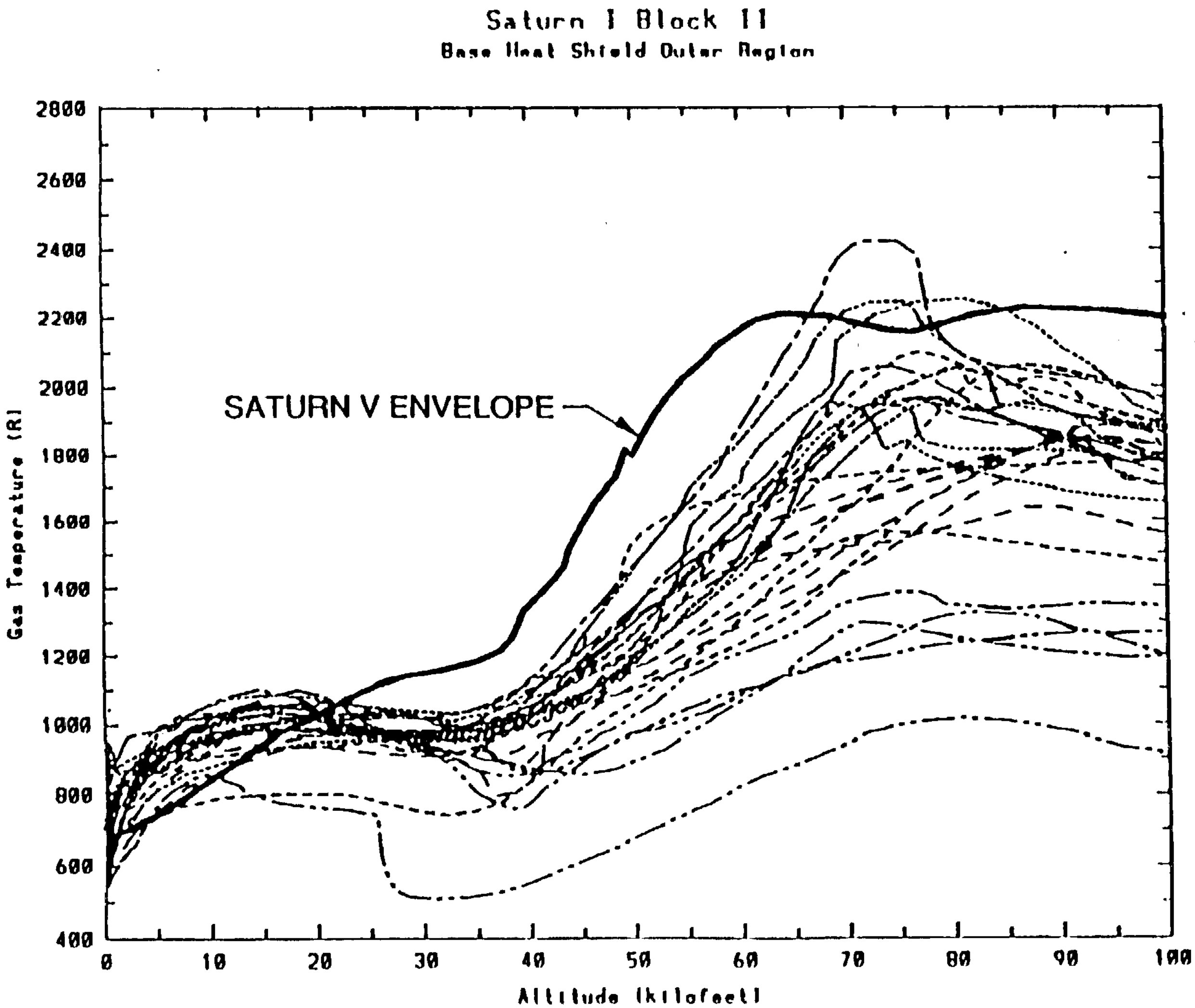
14 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK II INNER REGION

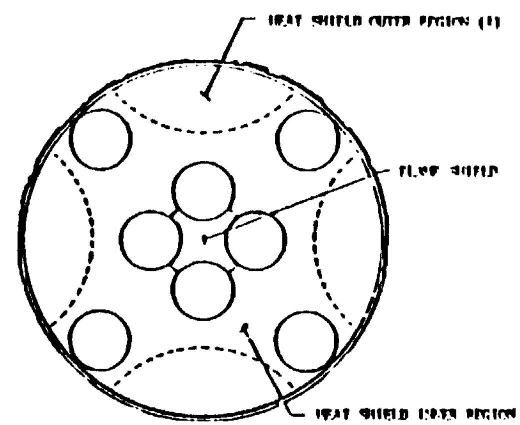




SATURN V GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN I, BLOCK II OUTER REGION

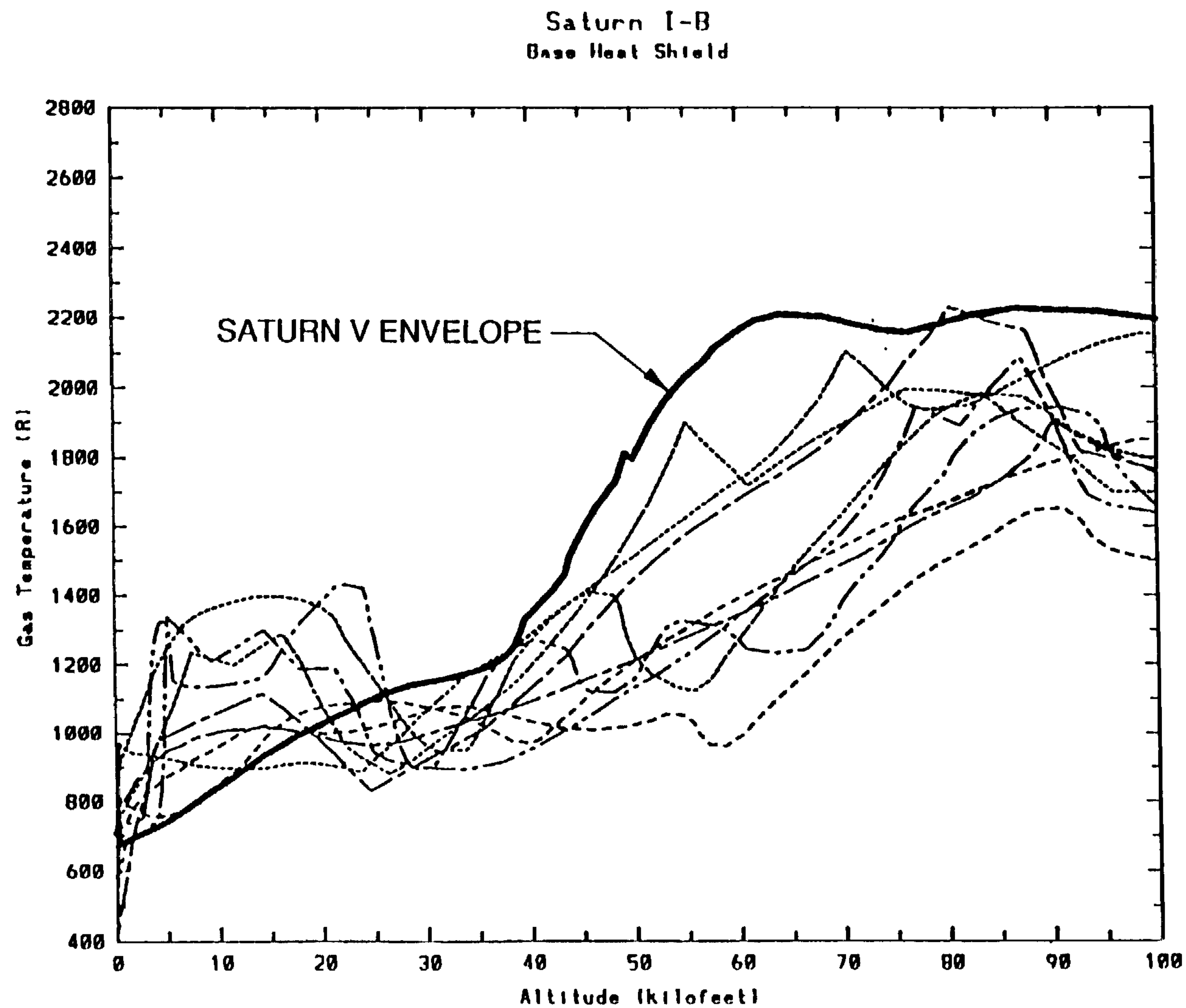
28 FLIGHT MEASUREMENTS FROM SATURN I, BLOCK II OUTER REGION



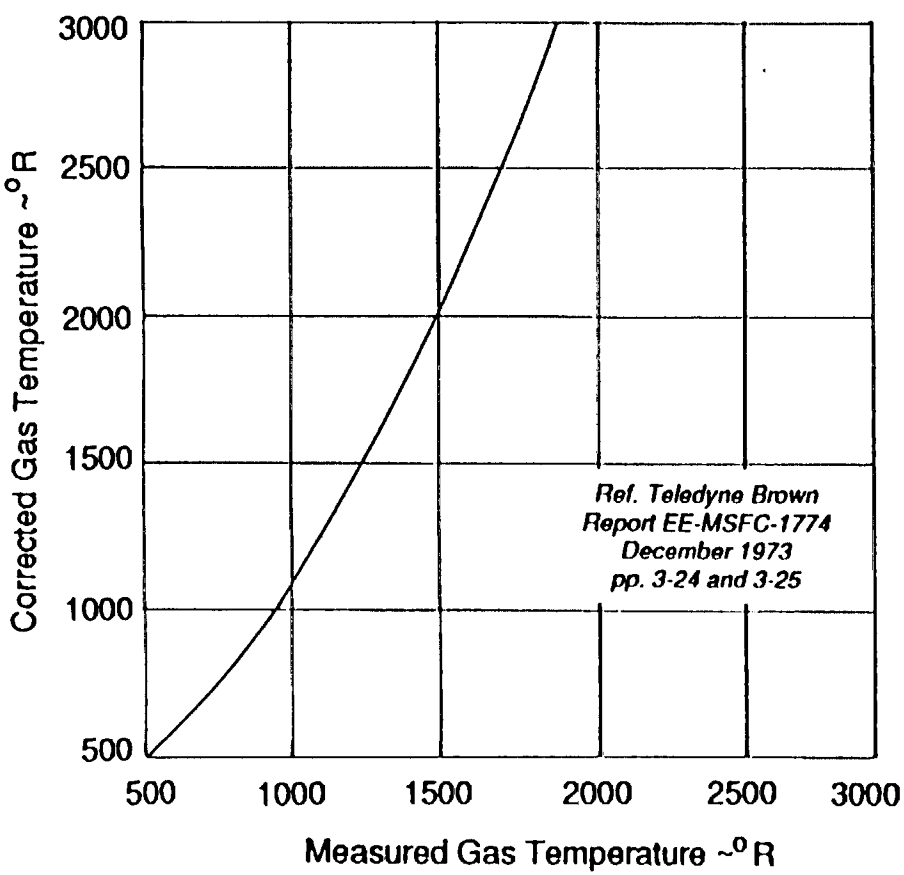


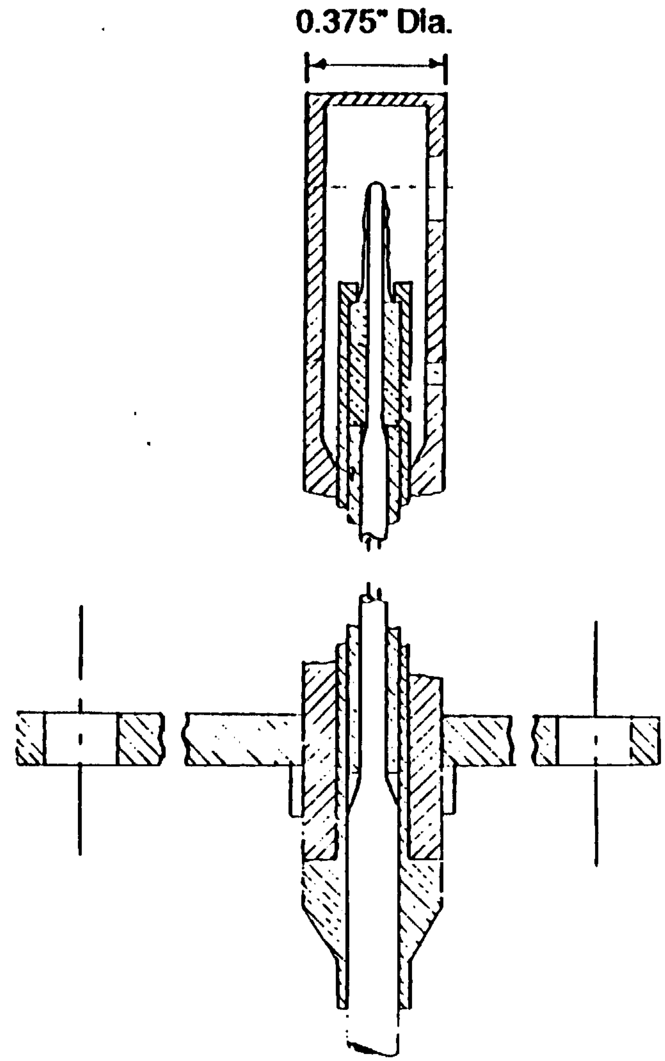
SATURN GAS TEMPERATURE BASE HEAT SHIELD
SATURN V vs SATURN IB

9 FLIGHT MEASUREMENTS FROM SATURN IB



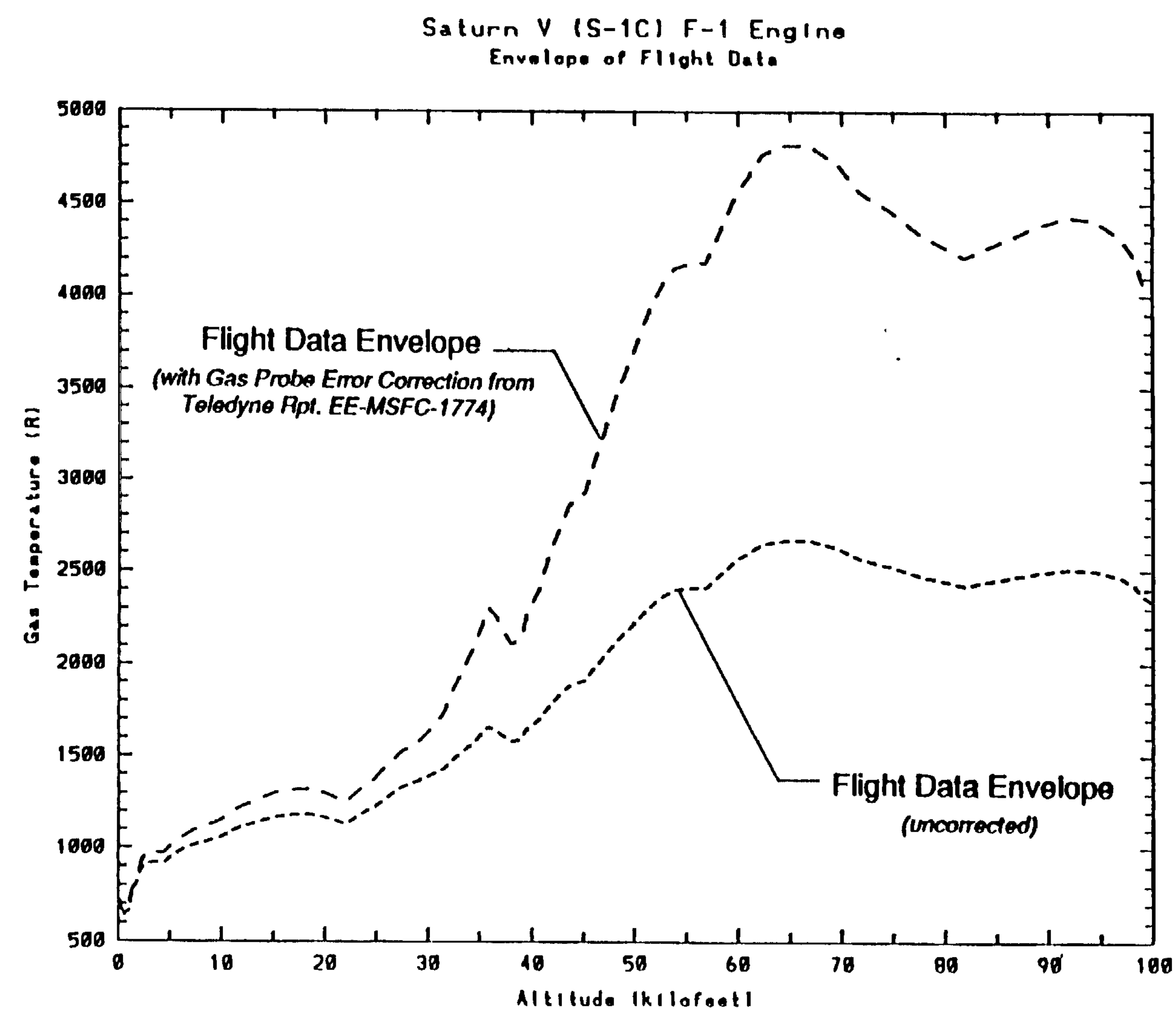
SATURN V S-IC STAGE GAS TEMPERATURE PROBE POST FLIGHT CALIBRATION





Saturn V S-IC Stage
Base Heat Shield Gas Probe

SATURN V F-1 ENGINE
GAS TEMPERATURE ENVELOPE



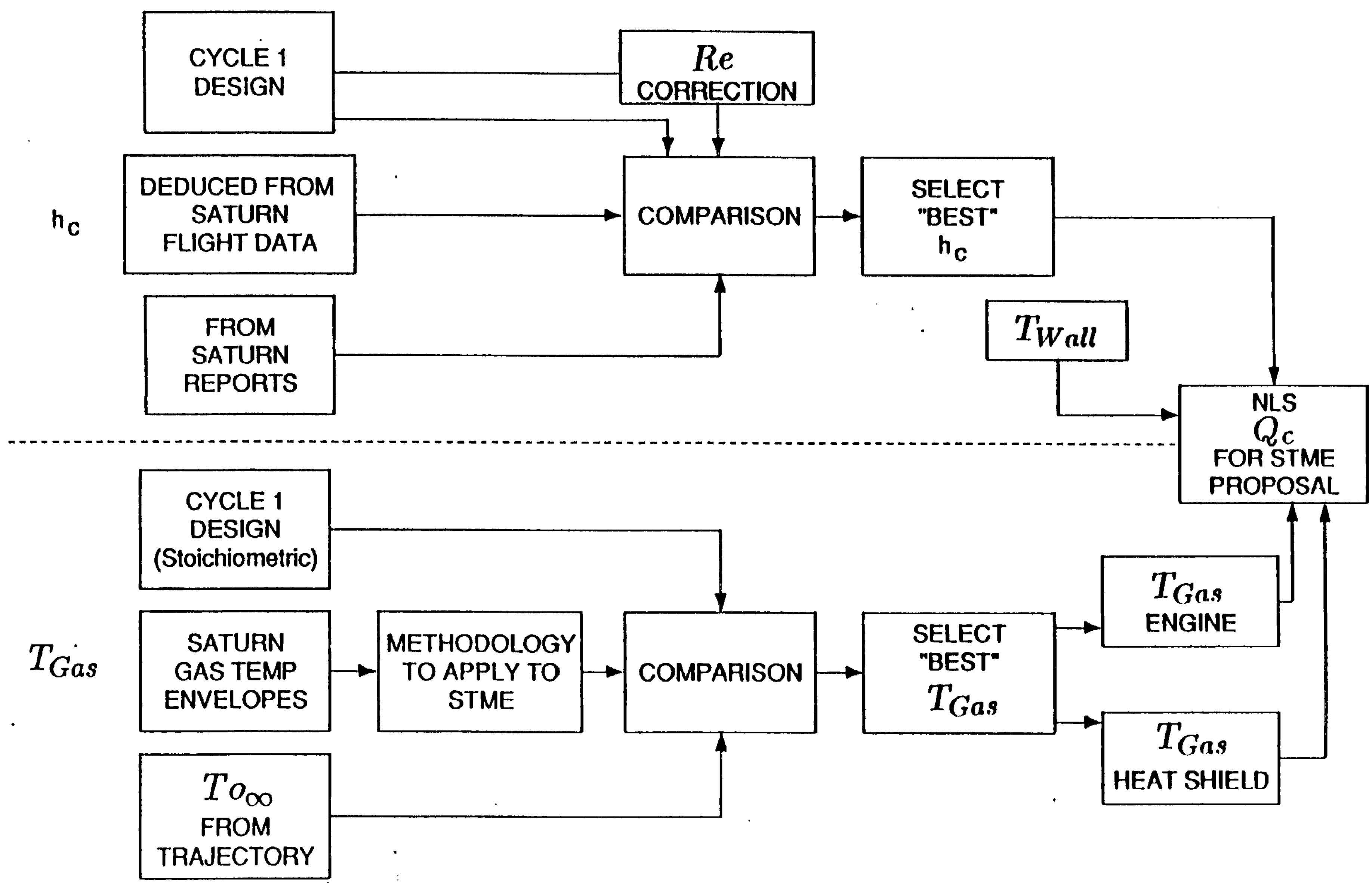
RESULTS OF SATURN V GAS TEMPERATURE REVIEW: OBJECTIVE 2

- The gas temperature data are relatively consistent and repeatable.
- Data have been separated into two main groups: base heat shield and inboard engine surfaces.
- AS-501 data are not included because of flow deflector effect.
- Envelopes of all data were determined as well as statistical mean and 1σ, 2σ, and 3σ standard deviations.
- Saturn V gas temperatures (excluding AS-501) are greater than freestream total temperature up to 100,000 feet - therefore, recirculation of hot gases had to occur!
- A technique to scale Saturn V data to NLS conditions was indicated.

NLS 1.5 STAGE
CONVECTIVE BASE HEATING METHODOLOGY

Engineering Approach based on Scaling
Saturn V Data to NLS Conditions

APPLICATION OF SATURN REVIEW RESULTS TO NLS 1.5 STAGE VEHICLE



SATURN V TO NLS 1.5 STAGE VEHICLE SCALING RATIONALE

BASIC ASSUMPTIONS:

$$\left(\frac{\dot{V}_{AIR}}{\dot{V}_{FUEL}}\right)_{SAT-V} = \left(\frac{\dot{V}_{AIR}}{\dot{V}_{FUEL}}\right)_{NLS}$$

Shear layer development/mixing along plume boundary assumed to be independent of turbine exhaust disposal scheme. Assume mixing differences driven by exhaust product density differences.

JUSTIFICATION:

Similar Base Geometry

- Engine spacing, total engine exit area/base area, engine length/base diameter

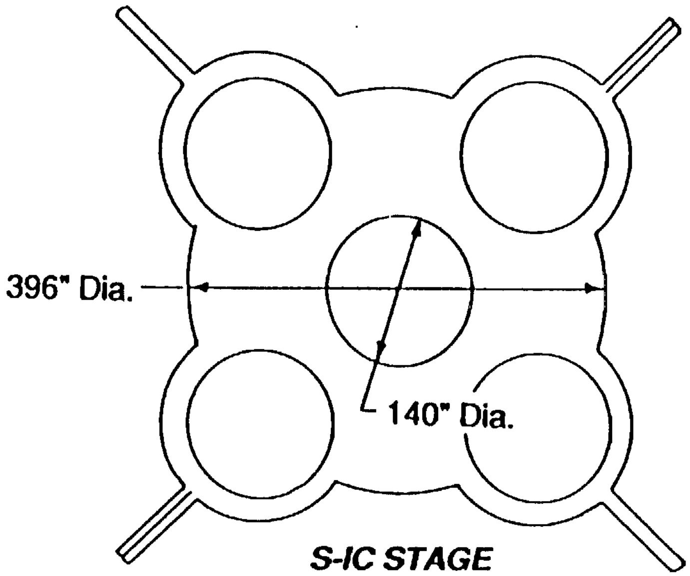
Similar External Flow

- Trajectories are comparable below 100 kft, freestream approach flow to plume boundary shear layer, and expansion into base similar

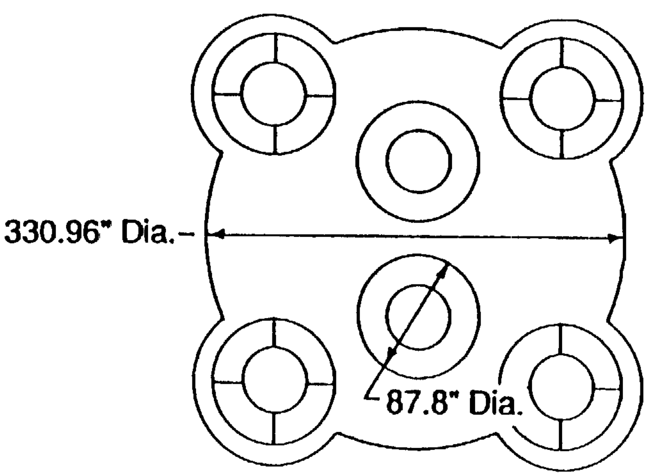
Turbine Exhaust Disposal Schemes

- Combustible turbine exhaust/total engine flow approximately equal, total turbine exhaust/total engine flow comparable

SIMILARITIES IN NLS 1.5 STAGE AND SATURN V S-IC STAGE



S-IC STAGE



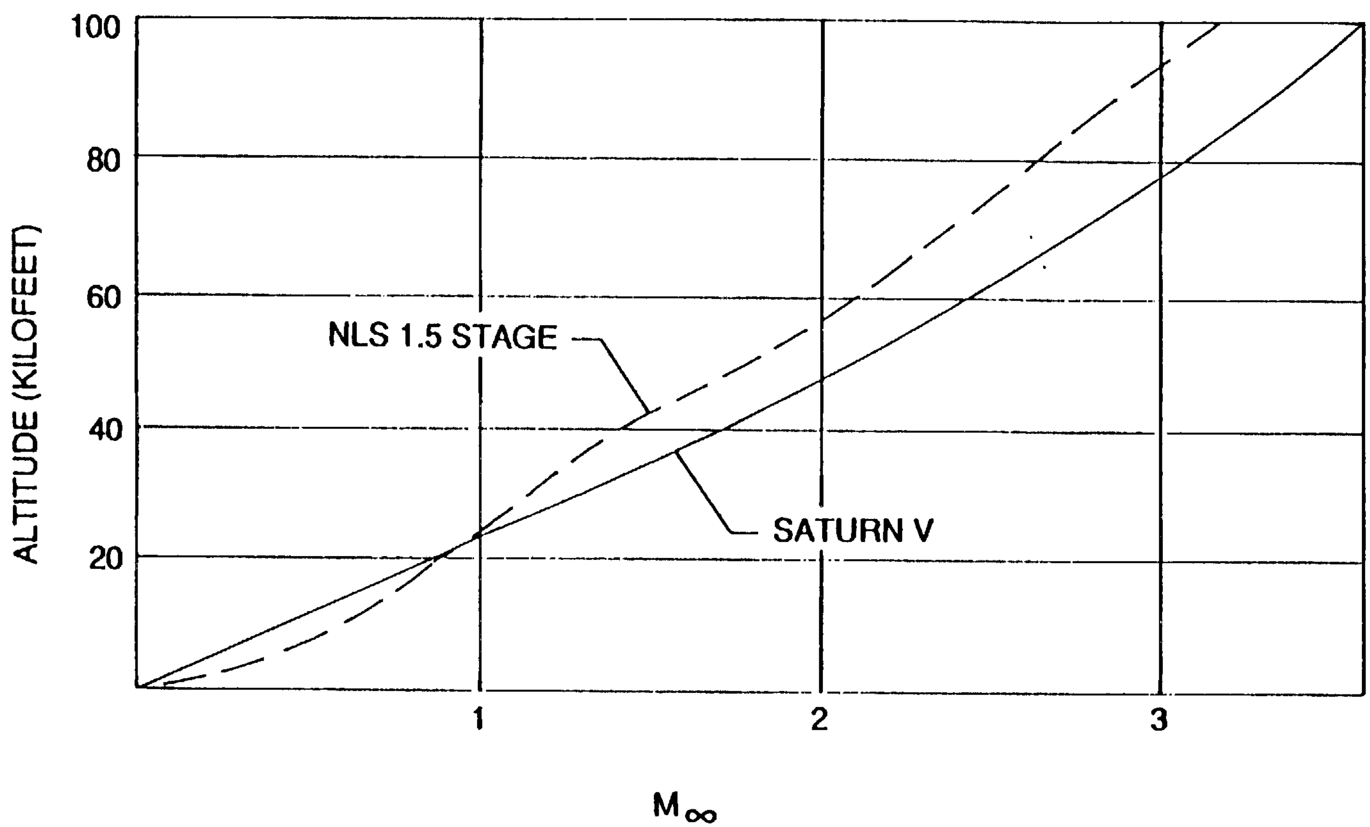
NLS 1.5 STAGE

Turbine Exhaust Composition

	F1 (lb/sec)	STME 580k (lb/sec)
H_2	13.40	30.11
CH_4	7.79	— —
CO	54.49	— —
C	69.31	— —
H_2O	10.80	33.95
CO_2	11.51	— —

	F-1 Engine	STME
Flow Rate		
Main Chamber, lbm/sec	5564.40	1292.70
Turbine Exhaust Total, lbm/sec	170.50	64.06
RATIOS		
$\frac{\text{Total Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0306	0.0496
$\frac{\text{Combustible Turbine Exhaust}}{\text{Total Engine Flow}}$	0.0216	0.0231
$\frac{\text{Total Eng. Exit Area}}{\text{Base Area}}$	0.6249	0.4223
$\frac{\text{Engine Length}}{\text{Base Diameter}}$	0.5742	0.4263
$\frac{\text{Nozzle Exit Diameter}}{\text{Engine Length}}$	0.6200	0.6209

NLS 1.5 STAGE AND SATURN V TRAJECTORY COMPARISON



METHODOLOGY FOR IMPROVING h_c EARLY IN FLIGHT

- Below 40,000 feet

BASE REGION REYNOLDS NUMBER LOW ALTITUDE CORRELATION

$$h_c \propto Re_B^{.8}\, Pr^{.33}$$ (Colburn Analogy for Turbulent Flow)

Steps:

1. Use velocity deduced from Saturn V pitot-static pressure data and compute $(\rho_B V_B)^{0.8}_{REF}$ at initial plume-to-plume recirculation assuming $P_B = P_\infty$ and $T_B =$ measured base gas temperature.

2. Compute ratio $\frac{(\rho_B V_B)^{0.8}}{(\rho_B V_B)^{0.8}_{REF}}$

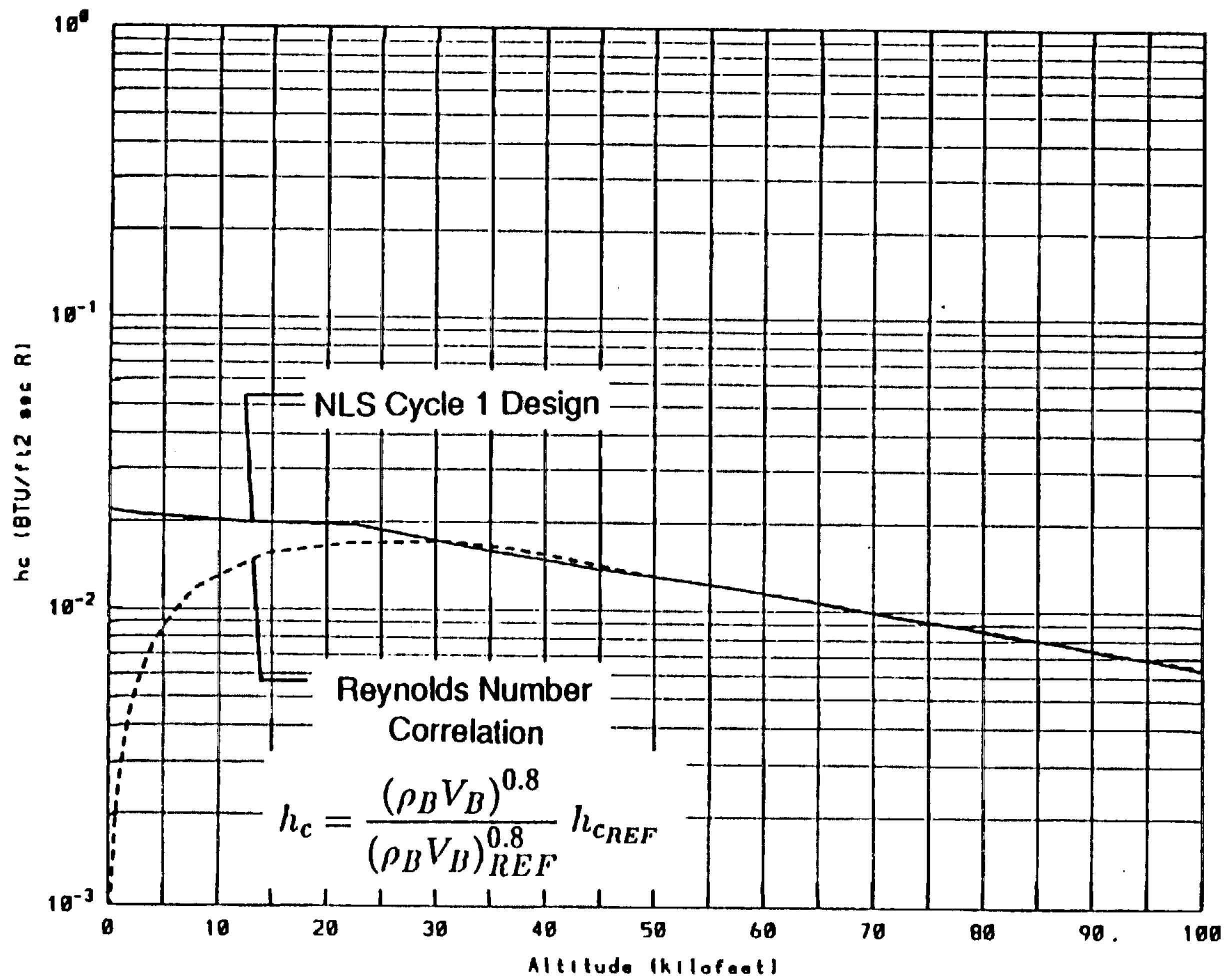
3. Then $h_c = \frac{(\rho_B V_B)^{0.8}}{(\rho_B V_B)^{0.8}_{REF}}\, h_{c_{REF}}$ for $0 \leq$ altitude $\leq$ 40 Kft

- Above 40,000 feet

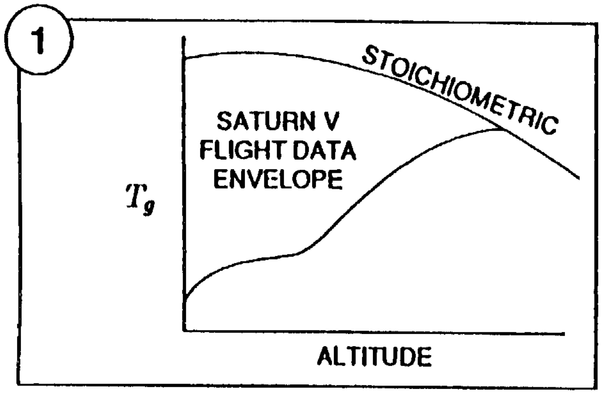
  h_c envelopes Saturn flight data

LOW ALTITUDE REYNOLDS NUMBER CORRELATION FOR h_c

STME Engine Nozzle



METHODOLOGY FOR APPLYING SATURN FLIGHT DATA TO NLS GAS TEMPERATURE

1


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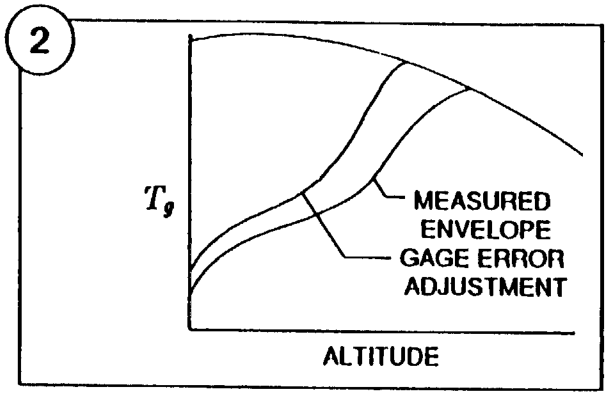

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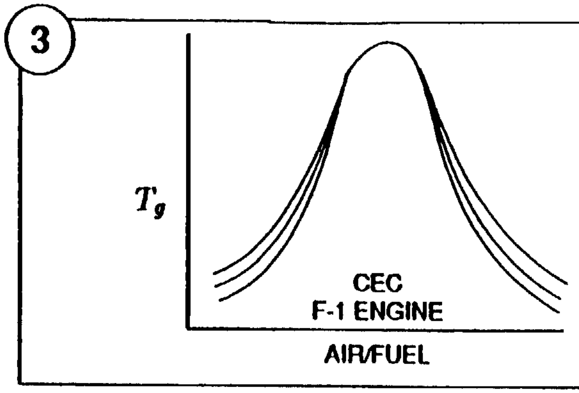

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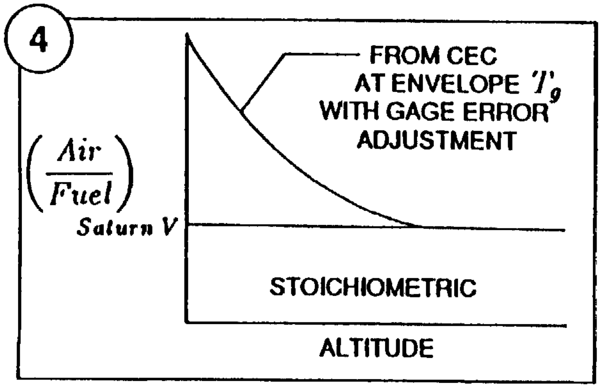
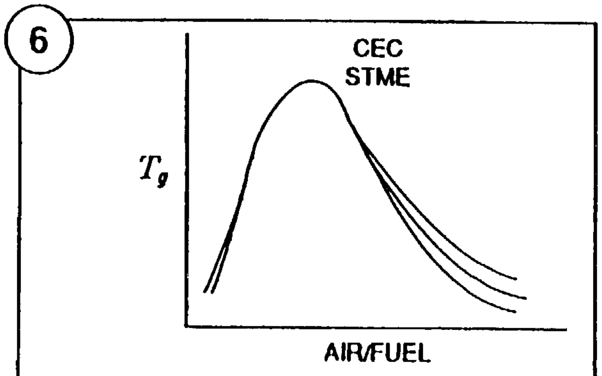


5

MASS ENTRAINMENT

- Assume NLS/Saturn V air/fuel volume flow rate ratios similar
- Then scaling relationship becomes:

$$\left(\frac{Air}{Fuel}\right)_{NLS} = \left(\frac{Air}{Fuel}\right)_{S}\left(\frac{\rho_S}{\rho_{NLS}}\right)$$

6


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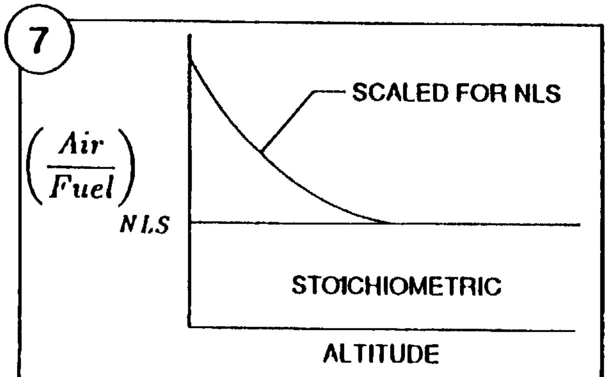

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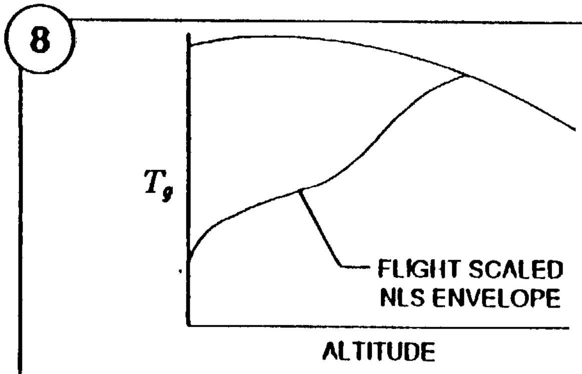
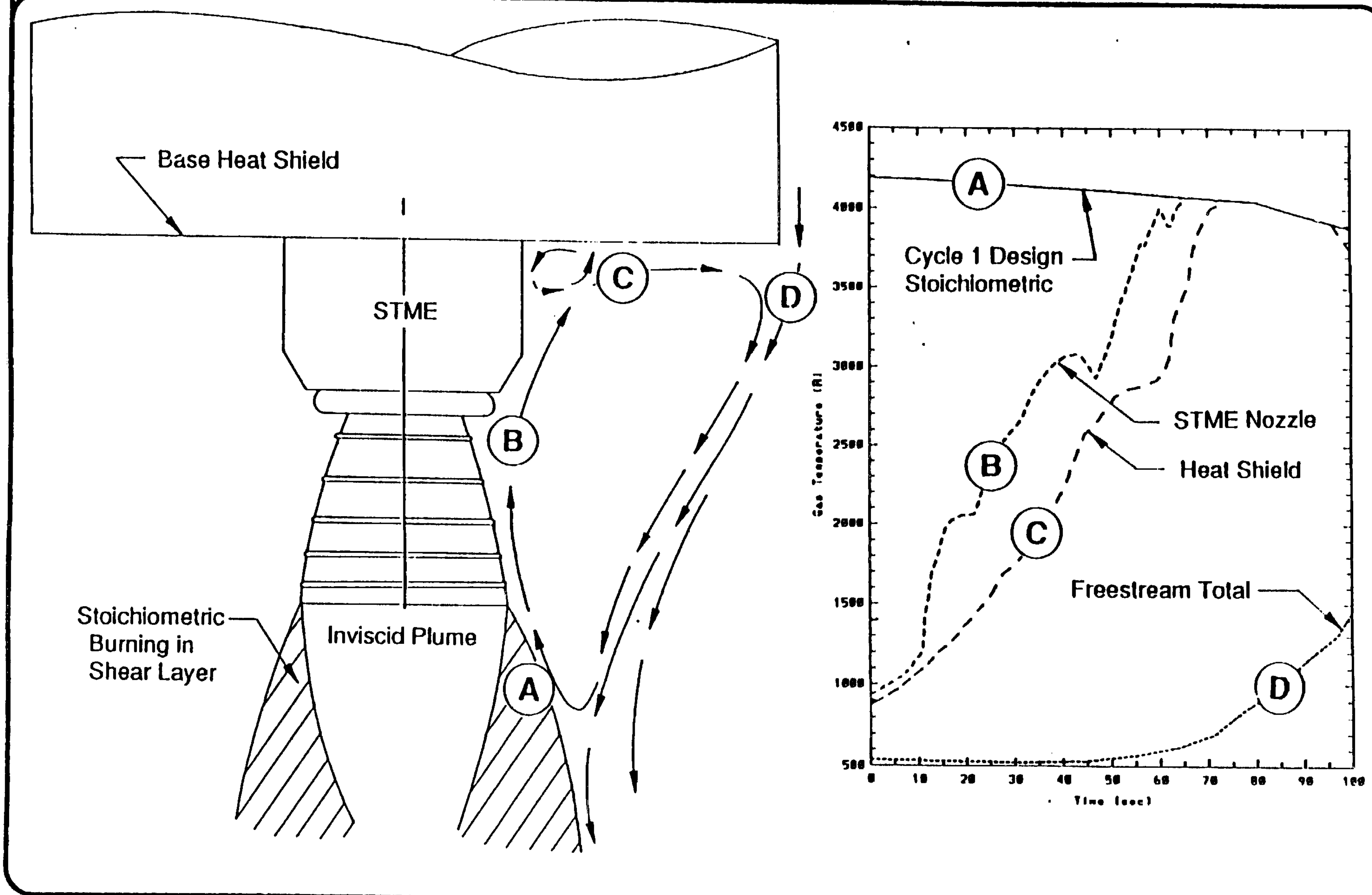
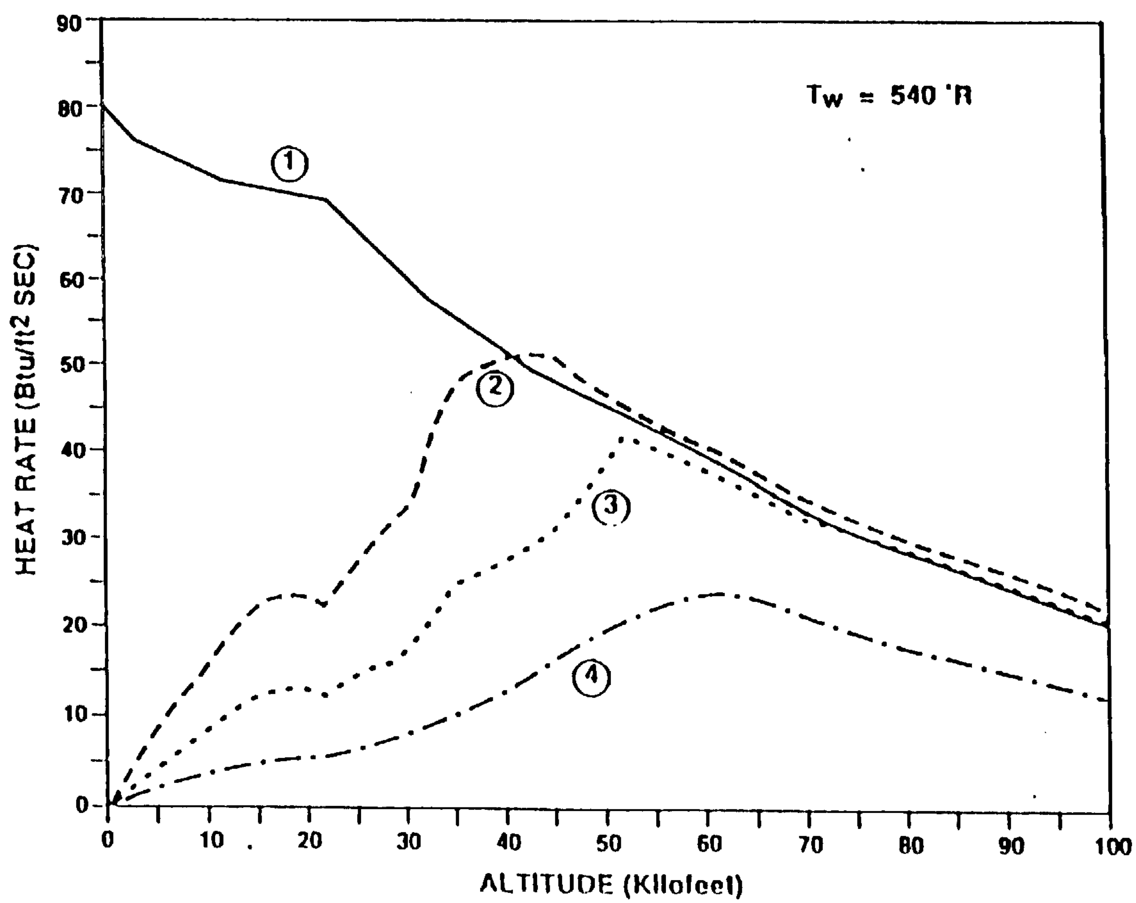


NLS SIMPLIFIED BASE REGION FLOWFIELD AT LOW ALTITUDES



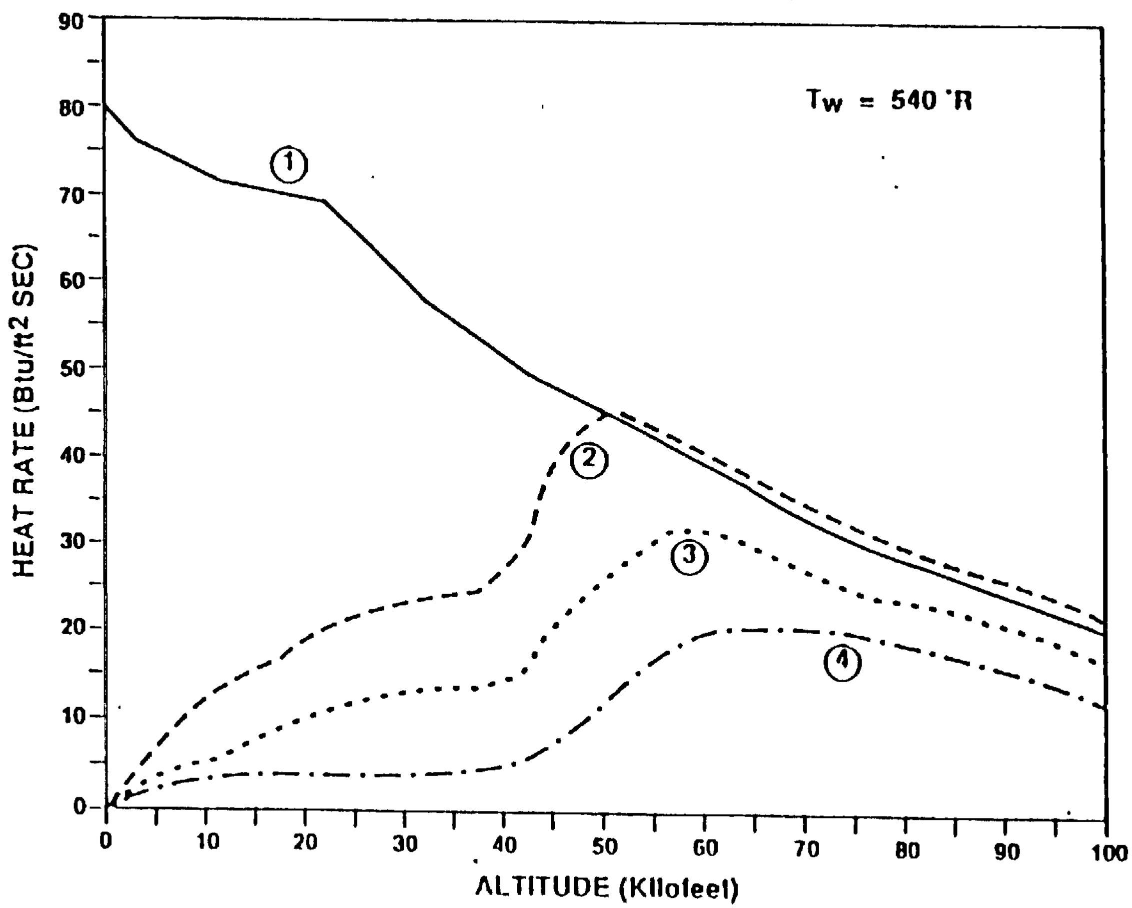
NLS STME NOZZLE CONVECTION ENVIRONMENT

CASE	DESCRIPTION	Q (Btu/ft^2)
1	CYCLE 1 – UPPER LIMIT	6,207
→ 2	C UNBURNED, h_c & T_{gas} ENVELOPED	2,617
3	C BURNED, h_c & T_{gas} ENVELOPED	1,861
4	C BURNED, h_c ENVELOPED & MEAN T_{gas}	984



NLS BASE HEAT SHIELD CONVECTION ENVIRONMENTS

CASE	DESCRIPTION	Q (Btu/ft^2)
1	CYCLE 1 – UPPER LIMIT	6,207
→ 2	C UNBURNED, h_c & T_{gas} ENVELOPED	2,144
3	C BURNED, h_c & T_{gas} ENVELOPED	1,356
4	C BURNED, h_c ENVELOPED & MEAN T_{gas}	736



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SENSITIVITY STUDIES AND SELECTION OF METHODOLOGY FOR THE CURRENT DESIGN ENVIRONMENT

REMTECH

SENSITIVITY STUDIES AND METHODOLOGY VERIFICATION

- Comparison (validation) of database - REMTECH vs. ED-64
- Different methods for enveloping Saturn flight data
- Sensitivity of h_c and q_c to choice of base region velocity
- Sensitivity of T_{gas} to entrainment adjustment $\rho_{RP-1}/\rho H_2$
- Sensitivity of T_{gas} to F-1 engine exhaust - carbon burning
- Comparison of convective heat loads with different assumptions

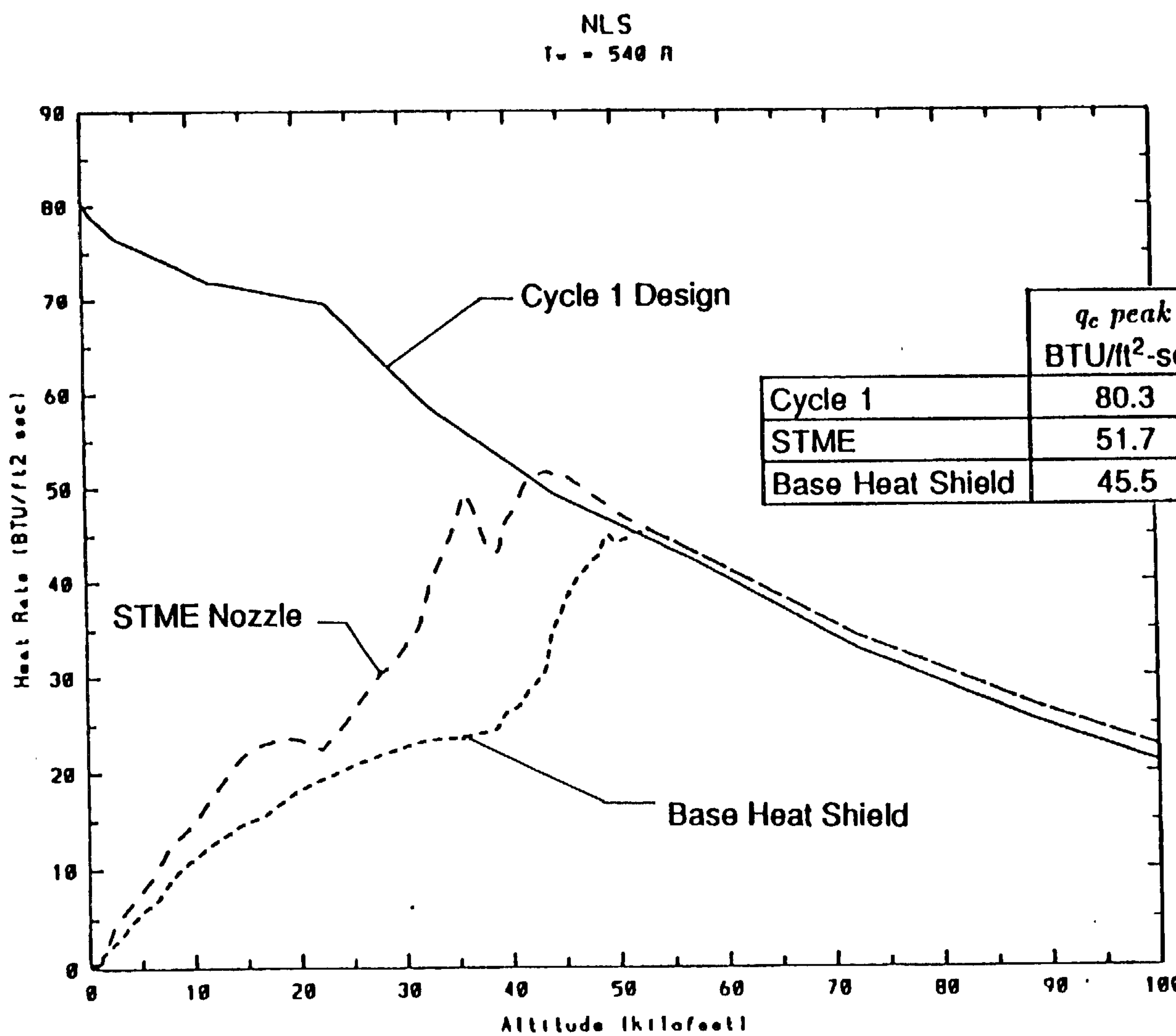
METHODOLOGY FOR CURRENT DESIGN ENVIRONMENT

- Gas temperature envelope based on F-1 engine flight data with gas probe error correction
- Air-fuel ratio from F-1 engine turbine exhaust (excluding carbon) combusted with air
- Shear layer entrainment scaling from F-1 to STME assumed similar after correction for turbine exhaust gas property and combustion differences
- Convective heat transfer coefficient based on:
  - a. Reynolds Number correlation from sea level to 40 Kft.
  - b. Envelope of Saturn data for altitudes above 40 Kft.

REMTECH

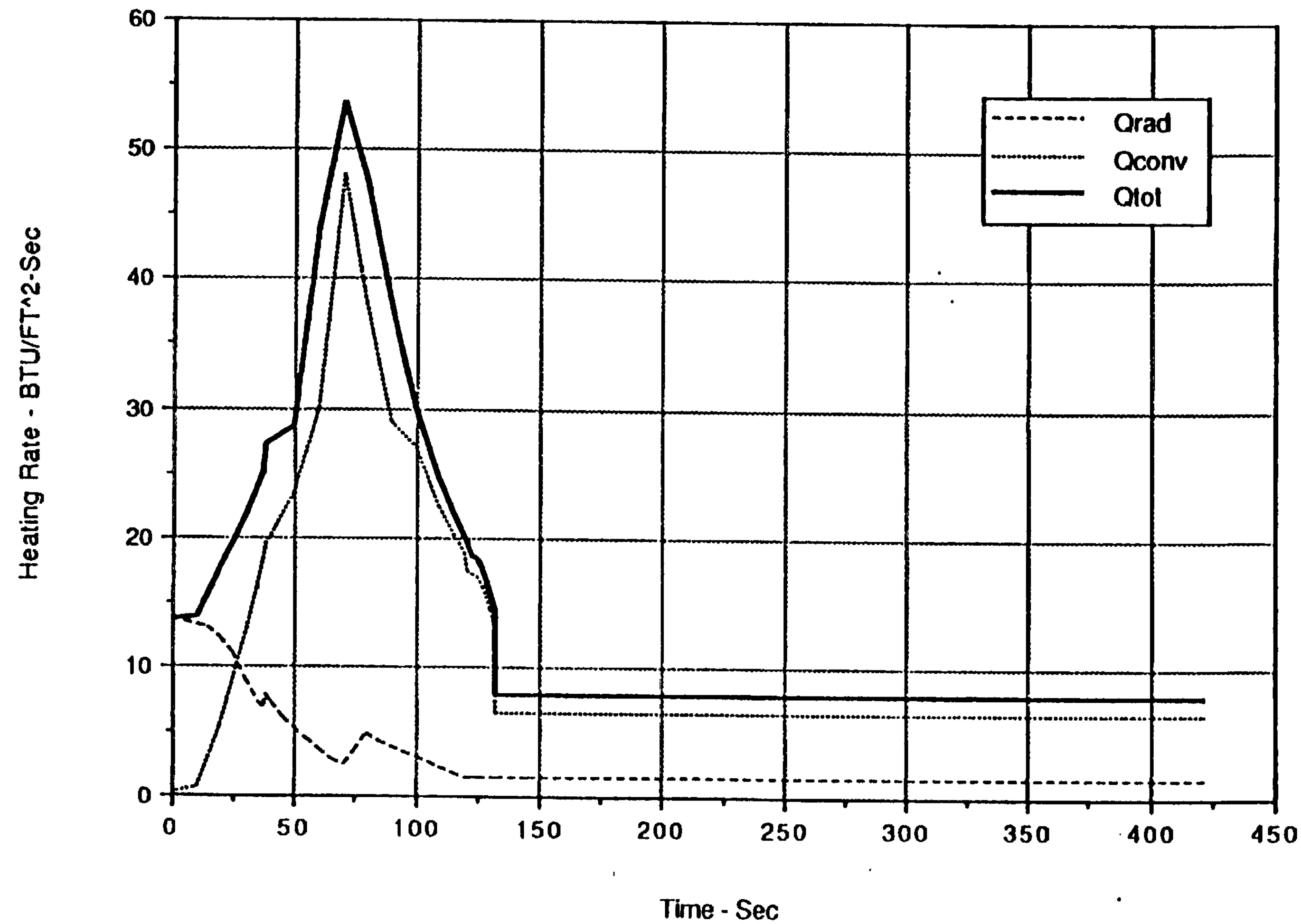
NLS 1.5 STAGE
CONVECTIVE BASE HEATING ENVIRONMENT

NLS 1.5 STAGE DESIGN CONVECTIVE BASE HEATING ENVIRONMENTS AT LOW ALTITUDES



	q_c peak BTU/ft^2-sec	Q_{load} BTU/ft^2
Cycle 1	80.3	6209
STME	51.7	2617
Base Heat Shield	45.5	2144

TOTAL BASE HEATING ENVIRONMENT FOR NLS 1.5 STAGE STME NOZZLE



CONCLUSIONS

- Convective environments determined by scaling Saturn V data to NLS conditions
- Environments substantially reduced when compared with Cycle 1
- Environments recommended by working group for NLS 1.5 stage design
- STME up-grade to 650 K should have minimal impact on environments